Laboratory Verification of Electric Double Layer Capacitor for Nanosatellites

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LABORATORY VERIFICATION OF ELECTRIC DOUBLE LAYER CAPACITOR FOR NANOSATELLITES

By

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The undersigned have examined the thesis entitled '**Laboratory verification of electric double layer capacitor for nanosatellites**' presented by **MUHAMMAD ALKALI**, a candidate for the degree of **Doctor of philosophy** and hereby certify that it is worthy of acceptance.

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EXECUTIVE SUMMARY

Electric Double Layer Capacitors (EDLCs) have a higher power density and lower storage capability in comparison to batteries. In terms of energy density, EDLC performance is currently between that of batteries and conventional dielectric capacitors. One of the main advantages of capacitors is their wide temperature range, whereas their main drawback is their low energy density. This thesis presents design considerations for the selection of EDLCs as energy storage components for nanosatellites. As nanosatellites development is driven by short development time and low cost, a simple EDLC power system was designed using commercial–off-the shelf (COTS) components. Ground tests, simulating real low earth orbit environment conditions, were performed. EDLC performances were analysed under the orbital simulated conditions. Testing demonstrated EDLC capability to be used as unit storage on nanosatellites under varied temperature, vacuum, radiation, long life and mechanical stress conditions. And the projected energy density showed that EDLC could be a substitute for the currently used space proven batteries in the near future. Also laboratory verification of EDLC based Power System for a Simple CubeSat Mission was performed for international space station (ISS) release was conducted. The thesis has five chapters

Chapter 1 gives an overview of space missions design and spacecraft subsystems. The EPS (electrical power system) is the livewire of the satellite which is responsible for generating power, regulating it, storing it for usage at peak, eclipse and crisis period, and distributes to the satellite. The design of these subsystems to meet mission objectives are translated to cost saving, lessened components, mass saving. Robustness is usually a big challenge in the process. For EPS the design revolves round the technology for solar array, energy storage such as battery, regulators and load distribution.

There have been several kind of batteries employed for spacecraft power systems. Selection of the batteries is determined by the depth of discharge (DoD), operating temperature range, safety, energy / power density and durability. Operations of such energy storage are highly controlled to survive the challenge of meeting the mission requirements.

EDLCs have gained prominence as emerging energy storage. EDLCs have very high power density, operable in wide temperature range (hot and cold temperature), high depth of discharge, multicycle capabilities. It has already gained prominence in terrestrial applications such as Hybrid electric vehicles (HEV) and Plugged- in electric vehicle (PEV). The apparent problem associated to EDLCs for space applications were enumerated thus; they have low energy densities (gravimetric and volumetric), no space heritage, low voltage level, COTS components developed for terrestrial applications

This chapter describes the aim of this study as seeing the applicability of EDLC as a replacement for the traditional batteries in electrical power system of spacecraft. The research aims at designing a simple power system that utilizes COTS components with lessened number of components. This will be useful especially in educational satellite projects that require less professional input, ease of handling, cheapness of parts procurement /development, shortness of incubation time.

Objective of this study is summarized as to

I. Perform a feasibility of COTS EDLCs to establish the applicability based on characterization test.

II. Develop a low and simple system using COTS components.

III. Perform laboratory verification of simple CubeSat mission using EDLC as energy storage.

Chapter 2 is on the literature survey of the EDLC. A brief background on the development of EDLC is given. The section also describes the materials and process involved in the construction of EDLC.

Chapter 3 points out the design requirement and considerations of energy storage for satellites. Series of ground based test of EDLCs for space application is conducted. EDLCs are promising as robust and easy-to-handle power storage components for nanosatellites. The disadvantage of small gravimetric and volumetric energy densities is constantly diminishing. It is only a matter of time until the energy densities become comparable to those of Ni-MH batteries. It is worthwhile to investigate the strength of EDLCs against the space environment. A series of environmental tests was carried out on a COTS EDLC.

The series of tests conducted include high-temperature (+65^{o}C), room-temperature, low-temperature (-25^{o}C, -35^{o}C) and vacuum tests under room temperature. Furthermore, total ionization dose, one-week duration, vibration and shock tests are conducted. EDLC post-durability functional check was conducted; in order to check for degradation. In the chapter, we set the judgment criteria of degradation to 10 % change of capacitance and 15% change of internal resistance between the virgin value measured on delivery and the final value after all the tests are done.

Chapter 4 attempts to verify the performance of EDLC as energy storage for CubeSat in ISS orbit by assuming and simulating very simple EPS topology for 1U CubeSat. Performance of EDLC as the sole power storage device is investigated when the average power requirement is less than 0.65W even without a charging regulator. At two different power profiles, confirmed by solar array simulator (SAS), performance of EDLC is checked. In the feasibility, comparison of EDLC based power system with the existing traditional battery in terms of total mass and simplicity is carried out.

Chapter 5 concludes the research with comments on future issues. EDLCs can effectively operate under wider operating temperature, in comparison to existing batteries. The performance of the capacitor is investigated under harsh space environments such as wide operating temperature (thermal cycle test), launch environment (vibration test), separation (shock test) and total ionization dose. Judgement criteria were set to ascertain the feasibility of the EDLC cell as effective energy storage for space. Scenarios for simple CubeSat mission are verified in the laboratory. The EDLC based power system was designed and implemented with lessened components (all components were COTS). The performance was efficient. The mass density of the capacitor based power system was higher in comparison with the existing traditional batteries-based power topologies. But volumetrically, EDLCs can fit into nanosatellite mission. With less regard for mass, EDLC based energy system is matured for use now and in the nearest future can be a replacement for batteries going by the energy density improvement based on the trend of development.

Keywords

Energy density, supercapacitor charge regulator, nanosatellite, irradiation

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DEDICATION

To Allah be the Glory.

This is dedicated to the four pillars of this pursuit: My late mother, my father, my own nuclear family and Cho-sensei.

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CHAPTER ONE: INTRODUCTION

Space missions design is composed of mission and objectives definitions. The architecture and the design of subsystems are critical in every mission.

A spacecraft subsystem is made up of the payload and bus system which include thermal & structure subsystem (TSS), on-board data handling subsystem (OBDH), electrical power subsystem (EPS), communication subsystems, Attitude and Orbit Control Subsystems (AOCS), Propulsion, etc. Each of the subsystem is saddled with responsibilities that together make the mission achievable.

The EPS is responsible for generating power, regulating it, storing it for usage at peak, eclipse and crisis period, and distributes to the satellite. The design of these subsystems to meet mission objectives which translates to cost saving, lessened components, mass saving and robustness is usually a big challenge in the process.

The EPS is composed of solar array, energy storage, battery, regulators and load distribution. There have been several kind of batteries employed for spacecraft power systems. The batteries used include Nickel based batteries such as NiH2, NiCd and NiMH or lithium ion batteries also referred as LIB.Selection of the batteries is determined by the depth of discharge (DoD), operating temperature range, safety, energy /power density and durability. Operations of such energy storage are highly controlled to survive the challenge of meeting the mission requirements.

Electrical Double Layer capacitors (EDLCs) widely known as supercapacitor have gained prominence as emerging energy storage based on some superior performance relative to traditional batteries.EDLCs have very high power density, operable in wide temperature range (hot and cold temperature), high depth of discharge, multicycle capabilities. It has already gained prominence in terrestrial applications such as Hybrid electric vehicles (HEV) and Plugged- in electric vehicle (PEV).

1.1 Problem Statement

The shortcomings for EDLCs in space applicability and applications over existing batteries include:

a. Low gravimetric energy density

b. Low volumetric energy density

c. Heritage

d. Low voltage level.

e. Non-space based Commercial off the shelf (COTS) components developed for terrestrial applications

Low gravimetric / volumetric energy densities of EDLC are quite a bottleneck in its utilization for EPS where dimension and mass are design consideration for most missions. In the nearest future this obstacle will be overcome with the current trend in capacitance, voltage and internal resistance enhancement.

EDLCs can be considered now and the nearest future as energy storage for spacecraft with some of the following attractive features:

- The ease of handling.
- High power density
- wide temperature operating range performance

A simple EDLC based power system devoid of complexity with lessened COTS components for simple missions is considered. Also space environments experiments to characterize the COTS grade EDLC were performed to check the survivability or durability of the cells for the existing harsh space environments cycle.

1.2 Research Aim, Objective and Approach

1.2.1 Aim

The aim of this study is to establish an EDLC based power system that can meet the power requirement of a nanosatellite mission. It is aimed at designing a simple power system that utilizes COTS components with minimal number of components. This will be useful especially in school satellite projects that require less of professional input, ease of handling, low cost of procurement /development, shortness of incubation time, safety, robustness and simplicity.

1.2.2 Objective

Objectives of this study are:

- To utilize different sized COTS EDLCs so as to establish its applicability based on characterization test.

- To expose sampled EDLCs to simulated space environmental conditions on the ground using the facilities at Kyushu institute of technology and a sister university.
- Setting experimental success criteria with regards to deterioration of capacitance and internal resistance of the EDLCs 10% and 15% respectively. This two parameters are key to performance of the capacitor.
- Perform laboratory verification of simple CubeSat mission using EDLC as energy storage.
- Design, develop and build the circuitry for the test completely from non-space based COTS components.

1.2.3 Thesis Organization

The organization of the thesis are as follows:

Chapter two outlines the fundamentals of NiMH, LIBs, EDLCs, as well as spacecraft power system architectures. Chapter 3 shows space environment test carried on EDLCs. In Chapter 4, shows laboratory emulation of a simple mission of EDLC based CubeSat power mission. The conclusion of the work and a list of suggestions for further work are presented in chapter five.

1.3 Author's selected Publications

<u>Journal papers</u>

Alkali, M., Edries, M.Y., Khan, A.R., Masui, H., Cho, M., "Design Considerations and Ground Testing of Electrical Double Layer Capacitor as an Energy Storage

Component for nanosatellite", Journal of Small Satellite (JoSS).[published in October issues, Vol. 04, No. 02 (October 2015) pp. 387-405].

Alkali, Edries, M.Y., Khan, A.R., Cho, M., ''Laboratory verification of Electric Double Layer Capacitor based power system for a simple CubeSat mission'' International Journal of Electrical Energy (IjoEE) [Published September issue in September issues, volume 3,number 3, pp. 122-129].

Conference papers

Alkali,M., Motohata,Tatsuo,S,Masui,H.,Cho,M. (2013):Supercapacitor: Testing it's practicability as power storage unit of a nanosatellite presented at the 5th Nanosatellite symposium 20-22, November 2013, Tokyo

Alkali,M.,Edries,M.Y.,Khan,A.R.,Masui,H.,Cho,M(2014):Performance evaluation of electric double layer capacitor as energy storage component of micro/nanosatellite on the imposition of varied temperature and vacuum conditions, presented at the 50th AIAA/ASME/SAE/ASEE Joint Propulsion conference 28-30 July 2014, Cleveland Convention Centre, Cleveland ,Ohio.

Alkali,M.,Edries,M.Y.,Khan,A.R.,Masui,H.,Cho,M(2014):Preliminary Study of Electric Double Layer Capacitor as an Energy Storage of Simple Nanosatellite Power System, presented at the 65th International Astronautic Congress (IAC), 29th September to 4th October 2014, Toronto, Canada.

Alkali,M.,Edries,M.Y.,Khan,A.R.,Masui,H.,Cho,M(2015)Environment Test Campaign of Commercial-off-the shelf Components of Electrical Double Layer Capacitor for space Use, present at the 34th ISAS/JAXA Space Power Symposium ,6th March,2015,

Institute of Space and Astronautical Science (ISAS)/Japan Aerospace Exploration Agency (JAXA) , Sagamihara, Japan

CHAPTER TWO: BACKGROUND AND LITERATURE REVIEW

2.1 Introduction

Batteries and electric double layer capacitors (EDLCs) are known to be energy storage components. In order to meet load requirement during eclipse time or when the load demands exceeds the generated energy at the primary source there is dire need for energy storage unit in spacecraft power system. This chapter features the existing batteries and EDLCs, their characteristics. The satellite power architecture is also briefly explained.

2.2 Background of nanosatellite

There have been high proliferation of nanosatellites mainly due to cheapness of development evident in short time of development and the use of commercial-off-the shelf (COTS) components. The term "Nanosatellite" sometimes shortened "nanosat" is an artificial satellite with ambiguous definition depending on who the stakeholder is. Nanosat has a wet mass between 1-10Kg according to European or American stakeholders but to the Japanese stakeholders, nanosatellites typically have masses and sizes less than 50Kg and 50cm x 50cm x 50cm respectively[Alkali,20013;Alkali,2014]. About 75 nanosatellites have been launched within ten years 10 years before 2014. But from November 2013 to January 2015 there was an outstanding jump rate of launches to 100 nanosatellites. Following the projection of 2014 by SpaceWorks about 410 to 543 micro/nanosatellites are expected to be launched globally by 2020[Buchen, 2014; Alkali, 2014].

Nanosatellites have high leverage in providing access to practical based education and thereby availing students the knowledge related to whole cycle of space projects, with gains in design evaluations, testing, etc. Also of note is the inherent lessons that are accruable from failures since project cost is low.It is also a means for testing new and innovative technologies without incurring heavy cost since there would be no need to spend so much on technologies that were still in developing stage as was previously the reverse was the case.

Nanosatellites subsystems which include onboard computers, structure and thermal, communication, electrical power subsystem and attitude control still trending in developments.

2.3 Electrical Power Subsystem (EPS)

Electrical power system is the nucleus of every spacecraft, it is the basic requirement for every satellite mission. Most of the early birds were lost due to failure of this critical subsystem.

2.3.1 Units of EPS

Generally, an electrical power system of a satellite is mainly composed of three units namely: the primary energy source or the generating source of energy (i.e solar arrays, primary batteries) and secondary energy sources (energy storage component), and power conditioning and distribution unit.

2.4 Energy Storage

This section explains batteries and electric double layer capacitors as energy storage components with lithium ion capacitor (LIC).

2.4.1 Batteries

Batteries are the most commonly used storage technology for space missions. Batteries store energy in electrochemical forms. Batteries are categorized into primary or irreversible i.e not reusable after full discharged because the electrochemical reaction is not reversible and secondary /reversible / rechargeable i.e reusable after full discharge since the electrochemical reaction can be reversed.

2.4.1.2 Nickel and lithium-based Batteries

The types of batteries commonly use in the space industry are nickel – based and lithium - based. Although the nickel-based batteries have lower specific energy /gravimetric energy density and energy density/ volumetric energy density than those of lithium-based batteries; the heritage of nickel types are more.

The commonly used nickel-based batteries in the industry include nickel-metal-hydride (NiMH), nickel –cadmium (NiCd) and nickel hydrogen (NiH_2).

Lithium ion batteries (LiB) are currently gaining prominence due to high performance in terms of volumetric/gravimetric energy densities. For effective utilization of LIB cell, it must be highly controlled to avoid thermal runaway. The control measures include operating at low temperature, shallow depth of discharge, and overcharging protection.

2.4.2 General Background of Electric Double Layer Capacitor

Electrical double layer capacitor (EDLC) also known as ultracapacitor is sometimes referred as supercapacitor. The first commercially available EDLCs then referred as supercapacitor were from a Japanese company, Nippon Electric Company (NEC).A typical EDLC stores electrical energy in the interface that lies between a solid electrode and an electrolyte [Endo, 2001; Rightmire,1966].

The traditional electrostatic capacitors that are utilized as another form of energy storage for minimal power applications in analogue circuits, or at most as a short-term memory backup supplies apparently due to low capacitance [Endo, 2001; Kotz,1999; Rightmire, 1966].

The EDLCs are becoming attractive with revolutionary enhancement of their capacitances due to increase in the surface area and reduced resistance of electrodes which increase the charge and energy storage. Another evident attributes is the high power and rate of power delivery capability [Endo, 2001].

EDLCs occupy high standing among energy storage devices available today. They have succeeded in narrowing the gap that hitherto exists between capacitors and batteries in terms of energy storage. They have higher energy density than traditional capacitors and much higher power density than the state of art batteries [Kotz,1999].

The aim of this thesis is to see how the attributes of EDLC can be used to replace or compliment the currently used batteries in spacecraft power system.

This chapter include historical brief on the development of EDLC, the state of art, the working of EDLC and the model of the capacitors.

2.4.2.1 Brief History of EDLC

The concept of double layer capacitance was first put fore in 1857 by Hermann von Helmholtz. It was first patented as EC, based on the double-layer capacitance structure in 1957 by General Electric Company .The capacitor was composed of porous carbon electrodes using the double layer capacitance mechanism for charging. The Standard Oil Company, Cleveland, Ohio (SOHIO) patented a device that stored energy in the double layer interface [Becker, 1957; Burke, 2000;]. Nippon Electric Company (NEC) of Japan licensed the technology from SOHIO and introduced the first EC products to the marketplace as memory back-up devices in computers in 1957 .In 1971, after NEC took over the license from SOHIO [Rightmire, 1966; Boos, 1970]. They re-presented a commercially available EDLC, rebranded as ‘’supercapacitor ‘’.Currently, there are many manufacturers of EDLC and lithium- ion based (super) capacitor acronym or known as LIC [Endo, 2001; Kotz,1999; Rightmire, 1966].

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2.4.2.2 EDLC Construction

This section describes the materials and process involved in the construction of EDLC.

2.4.2.2.1 Electrode

The electrode type is critical for determining the electrical property of EDLC .This is a surface process that greatly determines the characteristics of the capacitance. There are many types of electrodes that can be used for the construction of EDLC such as metal-oxides, conducting polymer, hybrid, carbon, etc. But carbon is commonly used as high surface are electrode material apparently due to its cheapness, availability and heritage. Treatment of activated carbon materials greatly influence the performance of the EDLC.The performance is enhanced due to the porosity of the structure of the electrode surface and it is also dependent on how accessible is the pores to the electrolyte. Pore size must be large enough to allow easy access to electrolyte ion.If the pore size are small, they may not allow mobility of ions. The pore size apparently are selected to fit the size of ions of the electrolyte [Burke, 2000; Endo, 2001; Kotz, 1999; Rightmire, 1966].

2.4.2.2.2 Electrolyte

The type of electrolyte used in making the EDLC is very important in the selection of the kind of electrode to be used .The magnitude of the energy density of the capacitor depends on the voltage level attainable, and the voltage level is dependent on the breakdown voltage of the electrolyte. The equivalent series resistance (ESR) is dependent on the electrolyte conductivity; the ESR affects the power density.

Organic electrolyte is more commonly used than the aqueous type in the construction of EDLC. The voltage level for organic electrolyte is as much as 2.5-3.0V.Organic electrolytes have higher resistivity with lower cell power while aqueous electrolytes have lower voltage breakdown as low as 1V but better conductivity than the organic type. The size of the capacitance is influenced by the electrolyte used. The more accessible is the electrolyte ions to the electrode's porous surface-area, the more charges that can be stored .In this case to achieve optimal performance, there must be optimal size of both the pores in the electrode and ions in the electrolyte [Burke, 2000;Endo, 2001; Rightmire, 1966].

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2.4.2.2.3 Separator

The separator is to electrically insulate the two electrodes, while ions are permeable; hence permitting the occurrence of ionic charge transfer. For the organic electrolytes the commonly used separators are papers or polymers while ceramics or glasses are commonly used with aqueous electrolytes. The attributes of a good separator for enhanced performance of EDLC include high thinness, high electrical resistance or high ionic conductance [Endo, 2001; Kotz,1999; Rightmire, 1966].

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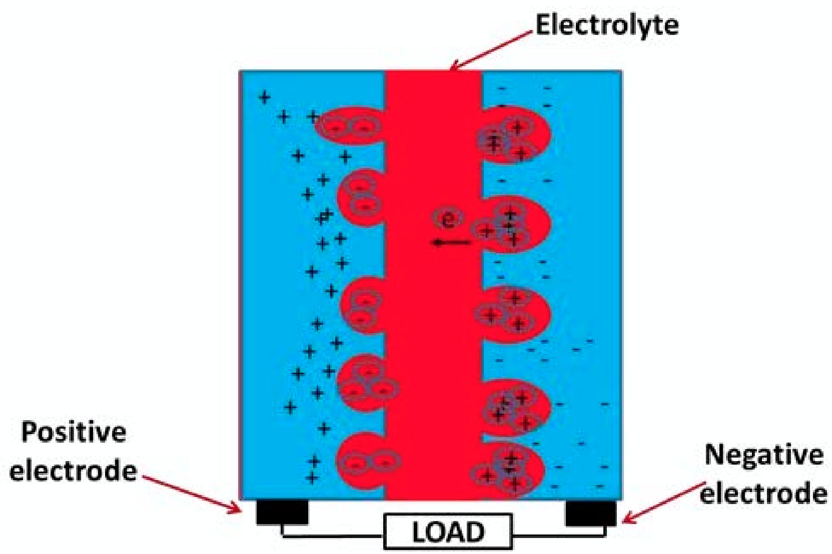

Figure 2:1 Structure of electric Double Layer Capacitor

2.4.2.3 EDLC circuit

The circuit schematic of EDLC is shown in figure 2:2. The EDLC is composed of four ideal elements namely; capacitance, C, equivalent parallel resistor, r_p, equivalent series resistance (r_s), and inductor in series (L). Notingly, the r_s contributes to loss in energy during discharging of the capacitor. The r_p sometimes referred to as leakage current resistance results in energy loss due to capacitor self-discharge. The r_p is much greater than r_s and is often neglected. For the inductor, it is usually small but cannot neglected.

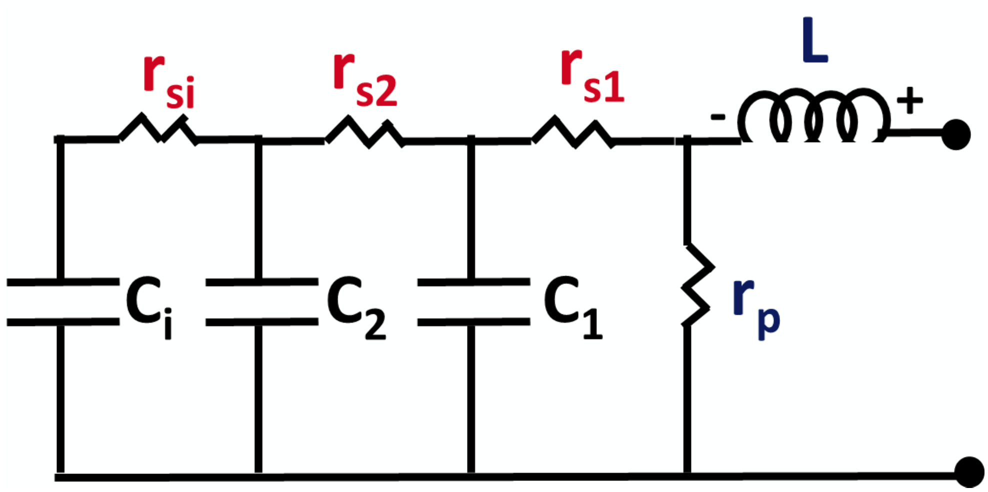

Figure 2:2 EDLC circuit schematic

2.4.3 Lithium ion Capacitor

The lithium ion capacitor (LIC) has combined traits of EDLC and lithium ion batteries with respect to their cathode and intercalation mechanisms respectively. The LIC has gravimetric energy density as much as 18Wh/kg and power density same as the EDLC. But the LIC is temperature and voltage regulated [Sivakkumar, et al 2012].

2.5 Missions with Supercapacitor based –power system

TurkSat-3USat a Turkish communications 3U satellite jointly developed by Istanbul Technical University, Türksat Company and Turkish Amateur Satellite Technology Organization (TAMSAT). TurkSat-3USat was launched on April 26, 2013[Wikipedia, portal 2015]. It is also a significant note that Opusat , a 1u educational cubest developed at the Osaka prefecture university was launched as a secondary payload on H-IIA 202 on 27th February,2014 had a hybrid power system with a lithium ion capacitor and lithium ion battery. The capacitor was to complement the demand the LIB and to demonstrate the LIC performance.

2.5 Reference

[1] Endo, M., Takeda, T., Kim, Y.J., Koshiba, K. & Ishii, K., "High power electric double layer capacitor (EDLC's); from operating principle to pore size control in advanced activated carbons," Carbon science, vol. 1, pp. 117-128, 2001.

[2] Rightmire, R.A., "Electrical energy storage apparatus", U.S. Patent 3288641, 29 Nov 1966.

[3] Kotz, R., & Carlen, M., "Principles and applications of electrochemical

capacitors", Electrochimica Acta, vol. 45, no. 15-16, pp. 2483-2498, 1999.

[4] Boos, D.L., "Electrolytic capacitor having carbon paste electrodes ", U.S. Patent 3536963, 27 Oct 1970.

[5] Becker, H.I., "Low voltage electrolytic capacitor ", U.S. Patent 2800616, 23 July 1957.

[6] M.Alkali, M.Y.Edries, A.R. Khan, H.Masui, M.Cho, "Preliminary Study of Electric Double Layer Capacitor as an Energy Storage of Simple Nanosatellite Power System", presented at 65th International Astronautic Congress (IAC), 29th September to 4th October 2014, Toronto, Canada.

[7] Burke, A., "Ultracapacitors: why, how and where is the technology," Journal of Power Sources, Vol. 91, Issue 1, 2000, Pages 37-50

[8] Alkali,M.,Edries,M.,Almubarak,H.,Khan,A.R.,Masui,H., Cho,M., "Performance evaluation of electric double layer capacitor as energy storage component of micro/nanosatellite on the imposition of varied temperature and vacuum conditions,'' 50th AIAA/ASME/SAE/ASEE Joint Propulsion conference 28-30 July 2014, Cleveland Convention Centre, Cleveland ,Ohio

[9] Buchen,E., "2014 Nano / Microsatellite Market Assessment," SpaceWorks Enterprises, Inc. (SEI) Atlanta, GA (2014), http://www.sei.aero/eng/papers/uploads/archive/SpaceWorks_Nano_Microsatellite_Market_Assessment_January_2014.pdf

[10] Nakasuka,S.,Kawashima, R. “Micro/Nano-satellite Activities by Japanese Universities and Vision towards International Contribution,” Committee on the Peaceful Uses of Outer Space: 55th session 6-15 June 2012

[11] Alkali,M., Edries, M.Y., Khan, A.R. , Masui, H., Cho, M.,: “Preliminary Study of Electric Double Layer Capacitor as an Energy Storage of Simple Nanosatellite Power System”, presented at 65th International Astronautic Congress (IAC), 29th September to 4th October 2014, Toronto, Canada.

[12] Becker, H.I., :“Low voltage electrolytic capacitor”, U.S. Patent 20800616, 23 July 1957

[13] Boos, D. L.,: U.S. Patent 3,536,963, “Electrolytic Capacitor Having Carbon Paste Electrodes” (Oct. 27, 1970).

[14] Sivakkumar, S.R.; Pandolfo, A.G. "Evaluation of lithium-ion capacitors assembled with pre-lithiated graphite anode and activated carbon cathode". Electrochimica Acta 65: 280–287. 2012-03-30. doi:10.1016/j.electacta.2012.01.076

[15] https://directory.eoportal.org/web/eoportal/satellite-missions/t/turksat-3usat assessed on 26th October,2015.

[16] https://en.wikipedia.org/wiki/TurkSat-3USat#cite_ref-itu1_2-0 accessed on 26th October,2015

CHAPTER THREE: DESIGN CONSIDERATIONS AND GROUND TESTING OF ELECTRIC DOUBLE CAPACITOR AS AN ENERGY STORAGE

3.1 Introduction

Every life form or vehicle is dependent on energy to function; satellites, just like other man-made systems, are non-functional boxes without electrical energy. The importance of an electrical power system (EPS) for the functionality of a spacecraft cannot be over-emphasized. The EPS of the satellite primarily functions to generate, store, and distribute electrical energy to the different sub-systems when they require it. The basic units of the subsystem include solar panels, which generate electricity from received solar energy; the power control unit, which is the regulation, and distribution unit; and the batteries, which are the storage units for backup in critical situations and/or during an eclipse. For the past fifty years, different kinds of batteries have been used to provide an effective and constant power supply for space missions during critical situations and/or eclipse periods. The lifespan of a mission is dependent on the lifetime of uninterruptible power supply, which is based on the number of life cycles of the storage unit. Rechargeable batteries that can operate for many cycles are, therefore, well suited for space missions (Hyder, 2003, Patel 2005).

The main energy storage systems currently used on board spacecraft are nickel-based batteries such as nickel-cadmium (NiCd), nickel-hydrogen (NiH_2), or nickel-metal hydride (NiMH) batteries, which are gaining popularity due to their outstanding performance, close to that of lithium-ion batteries (LIB) (Hyder, 2003, Patel,2005 , Ratnakumar et al 2003,Alkali et al., 2013,2014). In this research, a comparison is made between electric double-layer capacitors (EDLCs) and NiMH/NiCd batteries, because these are the most highly used storage devices for nanosatellites. NiH_2 batteries are mainly used for geostationary earth orbit (GEO) communication satellites, which fall out of the scope of this article. LIB batteries have a lower heritage compared to Ni-based batteries even though they have a higher energy density compared to the Ni-based batteries. Their major drawback is thermal runaway, which puts a burden on nanosatellite developers to assure their safety.

EDLCs are becoming of interest in the energy storage industry as standalones or as complements to existing batteries. One of the most attractive characteristics of EDLCs is their capability of directly storing power without conversions between chemical and electrical energy. EDLCs are safe, with no risk of explosion or ignition, since they use static electricity to store power, unlike batteries such as LIBs, which use chemicals whose reactions can change with temperature. The electrical power stored in EDLCs can be quickly supplied. This feature allows for quick discharge since the energy can easily be delivered and the reverse process makes it easy for the energy to be recovered. Moreover, the ratio of discharging time to charging time under the same current is almost equal to unit, i.e., 100%, which is greater than what is obtainable with batteries. EDLCs can undergo as many as 500,000 cycles and show performance degradation below 20%. EDLCs can perform effectively within a typical temperature range of -25^{o}C to 60^{o}C. The electrical charges are separated in the Helmholtz layers and the activated carbon electrodes have a large area, which provides EDLCs with low specific energy, low energy density, and high instantaneous power density. Recently developed EDLCs have energy densities as high as 7.4 Wh/kg, whereas Ni-based batteries can have energy densities as high as 60 Wh/kg (Alkali et al., 2014).

Nanosatellites are here defined as satellites with a mass typically between 1 and 10 kg. Commercial off-the-shelf (COTS) units are very attractive for nanosatellite development due to their short procurement time on the one hand and their low cost on the other hand. In 2013, 85 nanosatellites were launched, an increase of about 300% from 2012. From the projection analysis carried out in 2014 by Space Works, as many as 400 nanosatellites are expected to be launched by 2020 (Buchen, 2014). The importance of nanosatellites cannot be over-emphasized in the educational sector. From nanosatellite development, students gain practical experience in the whole cycle of a space-related project, which can be seen as professional training with such activities as design evaluation, testing, management, etc. Also of note are the inherent lessons that are accruable from failures, since the project cost is low. Nanosatellite systems suitable for education are desired. Such a system should be simple, easy to handle, safe and robust (Heidt, et al., 2000; Nakasuka, 2012).

The main function of spacecraft energy storage components is to maintain a constant power supply during eclipse periods, during which there are power cut-offs since the solar cells are not irradiated by sunlight, or in the event of anomalies. The main criteria for spacecraft energy storage selection are energy density, cost, multi-cycle capability, mass effectiveness, depth of discharge (DOD), round trip energy efficiency, vacuum-induced outgassing and sublimation effect of the storage material, voltage stability, ease in procurement and availability.

The energy density of a battery or any other storage component is dependent on the cell capacity, which is determined by fully discharging the cell to a minimum applicable voltage so as to maintain maximum performance retention without significantly degrading the characteristics of the component. Most energy storage components are conditioned to operate within certain DODs, temperature ranges, minimum ends of discharge, and current strain rates. Ni-based batteries have the greatest flight heritage, but preliminary testing presented in this paper demonstrates that EDLCs show promising performance as substitutes or complements of batteries for space applications, thanks to their DOD of 100%, wide temperature range, enhanced capacitance, operable under wide charging current (hence no need for strict current control), and low equivalent series resistance (ESR). Table 3.1 compares NiCd, NiMH, and EDLC energy storage performance.

For a feasibility case survey of EDLCs, we considered HORYU-II, a nanosatellite developed at the Kyushu Institute of Technology (KYUTECH). HORYU-II was piggyback launched by Japan Aerospace Exploration Agency (JAXA's) H-IIA rocket into polar Earth orbit of 680km on May 18, 2012. The satellite was equipped with nine AA rechargeable NiMH batteries, each with a capacity of 1900 mAh at a rated voltage of 1.2 V. The combination used by the satellite was three sets of three series- and three parallel- (3S3P) connected cells. The equivalent capacity, voltage and energy were 5700 mAh, 3.6 V and 20.52Wh respectively.

Table 3:1 Comparison of NiCd,NiMH and EDLC performance

Storage Type	**NiCd**	**NiMH**	**EDLC**
Rated voltage [V]	1.2	1.2	2.85
Gravimetric energy density [Wh/kg]	60-80	120	5-7.34
Volumetric energy density [Wh/l]	~60	120	~8
Gravimetric power density [W/kg]	150	500	13000
Operating DOD [%]	25	25	~100
Cost per Wh [¥]	245	300	1500
cycle life	1,000,000 or more	Upto 100,0000	Upto 100,0000
Operating temperature [°C]	0 ~ +45	0~ +45	-40 ~ +65
Leakage mitigation	Necessary	Necessary	Necessary
Charge rate	Limited	Limited	Unlimited
Outgassing	Positive effect	Positive effect	Positive effect
Overcharging mitigation	Strictly controlled -Temperature control -Voltage control -Charging current control	Strictly Controlled -Temperature control -Voltage control -Charging current control	Simple -Less concern for temperature. -Operating charging voltage below the rated value.
Heating issues while	No effect	No effect	No effect

recharging			
Memory effect	Yes	Yes	No
Self-discharge	Low	Lower	High

Table 3.2 shows the specification of state of the art commercially available EDLC cell that was used.

Table 3:2 Specification of a State-Of-Art Commercially Available EDLC cell

Commercial code mark	BCAP3400 P285
Rated voltage [V]	2.85
Capacitance [F]	3400
Equivalent series resistance (ESR) [mΩ]	0.28
Maximum usable voltage [V]	3
Gravimetric energy density [Wh/kg]	7.4
Volumetric energy density [Wh/l]	9.6
Rated operating temperature range [°C]	-40 to +65
Energy (Wh)	3.8

The potential energy was 20.52Wh, but as NiMH effectively operates at a DOD of 25%, the energy delivered was 5.13 Wh. HORYU-II's nine battery cells weighed 243 g (27 g per cell) and had a net gravimetric energy of 21Wh/kg. To have effective thermal control, the batteries needed to be accommodated and harnessed in a thermally and structurally fitted box. The total mass of the HORYU-II battery unit was 670 g and the usable gravimetric energy density with respect to the total mass (mass of batteries + mass of box + harness) was further reduced. Therefore, the net gravimetric energy density of

HORYU-II's battery unit was 7.66 Wh/kg with a net volumetric energy density of 20 Wh/l. HORYU-II's battery was charged by solar arrays with the open circuit voltage of

9V. The battery charging voltage was adjusted by inserting a charging regulator (DC/DC converter) between the solar array output and the battery charging input. A proper balancing between the operational point of solar array I-V curve and the battery charging is necessary to have an efficient system.

In comparison, EDLCs do not require any thermal control since they can withstand extreme space weather conditions and the gravimetric energy density is 7.4 Wh/kg, which corresponds to a volumetric energy density of 10 Wh/l. EDLCs also require a charging regulator to maximize the solar array power output especially because the EDLC voltage varies significantly assuming DOD is close to 100%. The charging regulator consumes little volume or mass. Therefore, for similar performance to the HORYU-II battery unit, an EDLC would be less complex and be a well-suited substitute in terms of gravimetric trade-off and thermal design. Figure 3.1 and Table 3.3 show a picture of the Horyu-II battery unit with the EDLC cell and their comparison profile, respectively.

When we make comparison between NiMH and EDLC, we should also look at the individual voltages and currents for the devices. HORYU-II bus voltage was regulated by the battery voltage, which was generated by 3 parallels of 3 series-connected batteries, which gave about 4 V during the nominal case. If the satellite bus is regulated by EDLC voltage, the maximum voltage is likely to be either 2.7 V or 5.4 V depending on whether EDLC is connected in series or not. The EDLC voltage can go down to very low as DOD would be very large. Power distribution device should accommodate such a wide variation. Power distribution circuit suitable for EDLC regulate bus is a subject of future study.

Figure 3:1 Battery Box of HORYU-II and EDLC cell

Table 3:3 Comparison of HORYU-II Energy Unit Configuration with EDLC

Energy Storage Unit	**Gravimetric Energy Density [Wh/kg]**	**Volumetric energy Density [Wh/l]**
State-of-art commercial EDLC	7.4	10
Horyu-II battery (3S3P) without box	21	49.25
Horyu-II battery unit with box	7.66	20

Figures 3.2 and 3.3 show EDLC gravimetric and volumetric energy density trends of development from 2004 to 2015 with projected energy density performance 10 years after. This is based on a survey of the capacitors that were on the market from 2004 to February 2015. According to the 10-year forecast, effectively by 2025, the energy density of a typical supercapacitor shall be gravimetrically increased up to 12.5 Wh/kg. The volumetric energy density shall increase up to 17.5 Wh/l by 2025 (Alkali et al., 2015). The numbers are good enough to consider EDLCs energy storage devices for nanosatellite power systems and LEO missions. This is especially true for educational

nanosatellites, which favour simple, robust and safe design over highly optimized cutting-edge design.

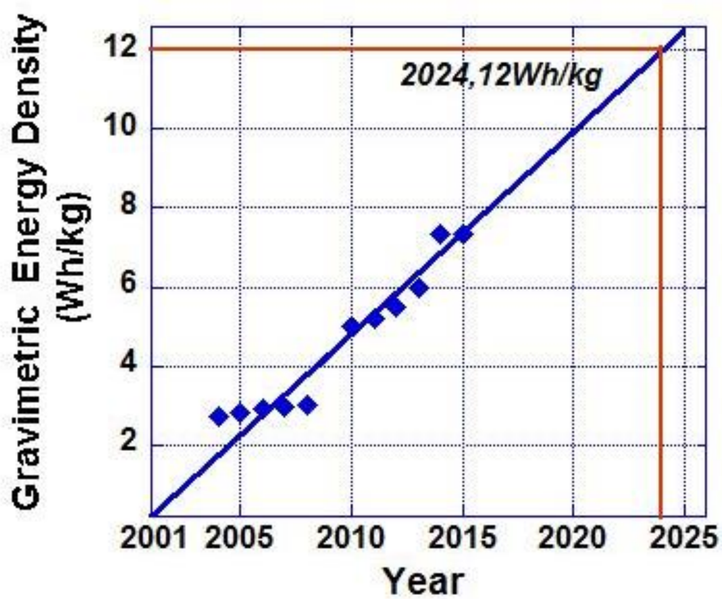

Figure 3:2 *Trend & forecast up to 2025 of EDLC gravimetric energy density*

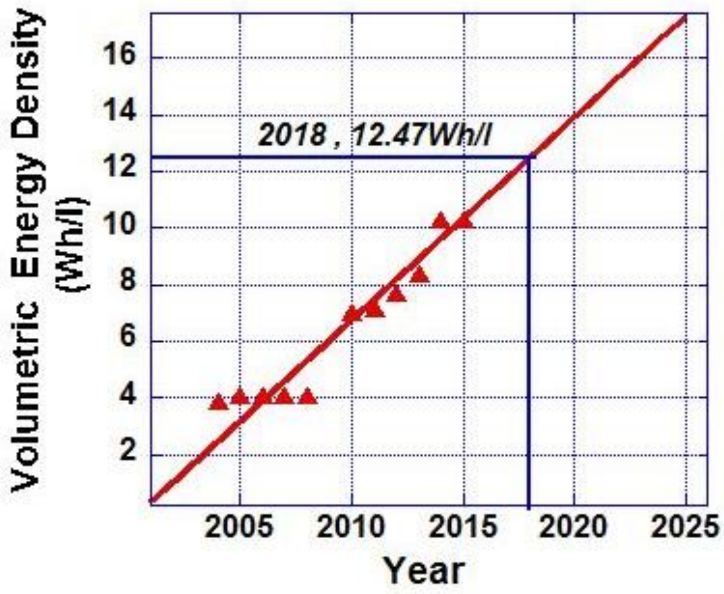

Figure 3:3 Tr*end & forecast up to 2025 of EDLC volumetric energy density*

To determine the strength of the EDLC to extreme LEO conditions, a test is carried out to subject the test article to varied temperature conditions, vacuum, and gamma ray irradiation. A simply designed supercapacitor charged regulator (SCR) was used for the implementation of a charging/discharging cycles test for EDLCs under various environmental conditions.

This chapter consists of four parts. The second part describes the experimental set-up. The third part discusses the experimental results. The fourth part concludes the paper with suggestions for future work.

3.1 Experimental Set Up

Table 3. 4 shows the specifications of the EDLC cell used for the tests. This EDLC cell is a COTS component and its development did not take into account its utilization in a

space environment. Throughout the entire test, no padding was done to partially or fully shield the EDLC from environmental damage. We did series of environment tests to one EDLC cell. During the test, the same EDLC was used. The success criteria are that EDLC functions properly in thermal and vacuum environment and no significant degradation occurred after exposure to various environmental factors. In the present paper, we set the judgment criteria of degradation to 10 % change of capacitance and 15% change of internal resistance between the virgin value measured on delivery and the final value after all the tests are done.

Table 3:4 Specifications of the EDLC

EDLC product name	**JJD0E408MSED**
Rated Voltage [V]	2.5
Capacitance [F] as manufacture's rated value	4000
Capacitance [F] measured on delivery	4183
ESR [mΩ]	2.2
Maximum Usable Voltage [V]	3
Specific Energy Density [Wh/kg]	3.5
Rated Operating Temperature Range [°C]	-25~60

Driven by the philosophy of simplicity and low cost, a SCR unit was developed using COTS components. The SCR was designed to extract the maximum power available from solar panels with high efficiency in conversion. The electrical scheme for the SCR operation is simple and includes an EDLC as its energy storage unit (test specimen), an electronic load, and a power source that simulates solar cell power generation. The SCR performs instrumentation of the electrical energy supply from the power source to the load, and then to the EDLC for its charge. An SCR circuit diagram is shown in figure 3.4.

The SCR contains an adjustable power module (PM), which is a buck converter (Texas Instruments), PTH08T230WAD. The buck converter was set to constantly provide a voltage of 3.42 V with an adjustable resistor of about 1.1 kΩ. To prevent current backflow from the EDLC, a diode was positioned after the PM. Current backflow could especially occur if the module stops supplying power, i.e., during discharge time (eclipse simulated time). Moreover, it also ensures a constant power supply to the load (satellite bus). The capacitors were included to reduce ripple voltages and the two 0.1-Ω-rated resistors served as measurement points for the DC and EDLC voltage and currents.

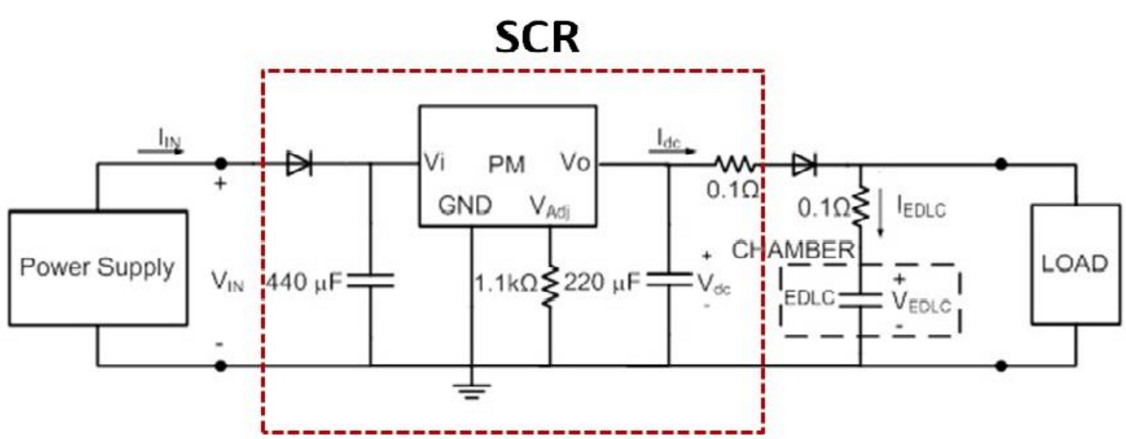

Figure 3:4. *Schematic diagram for the test in atmosphere/vacuum.*

A multi-range DC power supply (PS) (GW-INSTEK PSW 80-13.5) was used to simulate the solar cells and power profile. The PS was connected to the SCR to serve as power input to the power module, the power control unit supplied power to the EDLC, and the electronic load (PLZ164WA, KIKUSUI) served as the satellite load. For data acquisition, to simulate a LEO environment with an orbital period of 100 min (65 min sunlight and 35 min eclipse), the electrical energy supply/cut-off from the DC PS was controlled by the LabVIEW program. The 100 minutes cycles was used to simplify the 98 minutes cycle (63 min sunlight and 35 min eclipse) of HORYU-II which was launched to 680km.

NI DAQ instruments were used to observe the charge/discharge cycles. With this test setup, the desired charge/discharge cycling of the EDLC could be performed.

Besides controlling the PS, the program has a selecting feature for the desired load scheme (i.e., constant power, constant current or constant resistance). The program also collates and saves the measuring data of the PS, load, and the EDLC cell under test. For temperature variations (high and low temperature conditions), tests were implemented using a small thermostatic chamber (900 series – 925E, Despatch Industries). After testing under atmospheric conditions, the EDLC cell was then exposed to vacuum conditions (10^{-5} Pa) at room temperature using a vacuum chamber. An overview of the test conditions is given in Table 3.5.

Table 3:5 EDLC's Charging/Discharge Test Conditions

Operation in atmosphere			
Mode		**Period [minutes]**	
PS supplies power (sunlight simulation)		65	
PS cut-off supply (eclipse Simulation)		35	
Each cycle		100	
Various operation condition			
EDLC temp. [°C]	**Load**	**No. of Cycles**	**Environment**
65	Constant power (CP, 1.2W)	15	Thermostatic Chamber (TC)
-25	Constant power (CP, 1.2W)	15	TC
-35	Constant power (CP, 1.2W)	15	TC
Room temperature	Constant current (CC, 1A)	15	Vacuum Chamber
Room temperature	Constant current (CC, 1A)	9	Atmosphere
Room temperature	Constant current (CC, 1A)	1	Gamma radiation

The PS was conditioned to provide the input (i.e., voltage in (V_{IN}) and current in (I_{IN})) to the adjustable buck-converting PM. The 400-μF capacitance filters any eventual rippled voltage from the PS. The 1.1-KΩ resistor sets the output voltage, V_{dc}, to a value of 3.42 V. The DC current, I_{dc}, value is obtained using the drop from the first 0.1 Ω after the $V_{dc.}$ The diode was positioned to prevent the reverse flow of current due to EDLC discharging during eclipse time. The 0.1-Ω resistor before the EDLC cell is used to measure I_{EDLC}, which is the current flow in charging of the EDLC or the EDLC cell discharge current to the load during eclipse, i.e., when the PS was cut off.

In the first 35 minutes the power supply was in off mode, since the EDLC was not being charged and it was connected to the load. Immediately after the 35th minute of discharge time, the power supply was turned on by the LabVIEW program and delivers input power to the SCR for 65 minutes as well as to charge the EDLC. The energy supplied to the SCR was then buck-converted as V_{DC}. During this period, the capacitor was charged while delivering 1.2W power to the load. The choice of 1.2W was to emulate load delivery of 1W with the assumption of converters between the load and the EDLC during eclipse delivery. The power level of 1W assumes a typical power consumption of nanosatellites. Most converters have efficiency of 80%. Knowingly, during charging period the delivered dc current was divided so as to charge the capacitor and also to deliver to the load.

The power profile we employed in the present paper is very basic, which do not reflect the realistic operation in space such as occasional high power consumption due to communication or mission. In the paper we intended to answer the fundamental question whether EDLC can be used for space environment onboard nanosatellites. Therefore simple power profiles, constant power or constant current, were chosen. More realistic power profile will be tested in near future.

A thermostatic chamber was used [Figure 3.5 and Table 3.6 show the picture and specifications, respectively] to create high and low temperature conditions in atmosphere in order to check the performance of the EDLC during and after imposition. Before the cell was inserted in the thermostatic chamber, it was wrapped in a silver sheet and nitrogen gas was injected to free the wrap from air. The latter was performed to prevent condensation. Thermocouple sensors were attached to the EDLC cell to measure the temperature on the EDLC wall using a temperature DAQ controlled by the LabVIEW program. Figure 3.6 shows the test schematic when EDLC cell was set inside the thermostatic chamber for testing.

Figure 3:5 *Photograph of thermostatic chamber (EDLC was placed inside)*

Table 3:6 Thermostatic chamber specifications

Item	Specification	
Chamber	900 series – 925E-1-4-0-120	Manufacturer: Despatch Industries
	Test volume	0.02 m3
	Temperature range	-190˚C to 200˚C
	Power	2.392 kVA
	Heater	2000 W at 17.4 A
Data Acquisition System	DAQ NI 9213	
Data measurement software	LabVIEW	
Data Thermocouple type	K-Type	
Control Thermocouple type	T-Type	

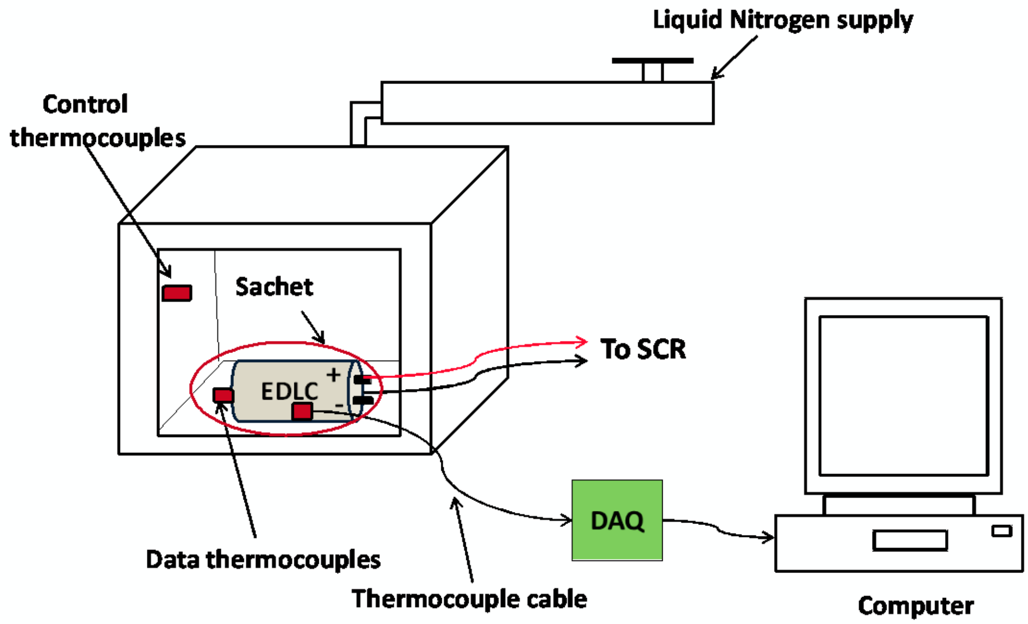

Figure 3:6 *Schematic of thermal cycle test setup.*

The thermostatic chamber was set to create a high temperature of 65°C or a low temperature of -25°C and -35°C, during which the charge/discharge tests were performed under the soak condition. Figure 3.8 shows the test bench for varying EDLC temperatures under atmospheric conditions. The LEO environment vacuum condition at room temperature was created using a small vacuum chamber (see Figure 3.9 for the test schematic and Table 3.7 for the chamber specifications).

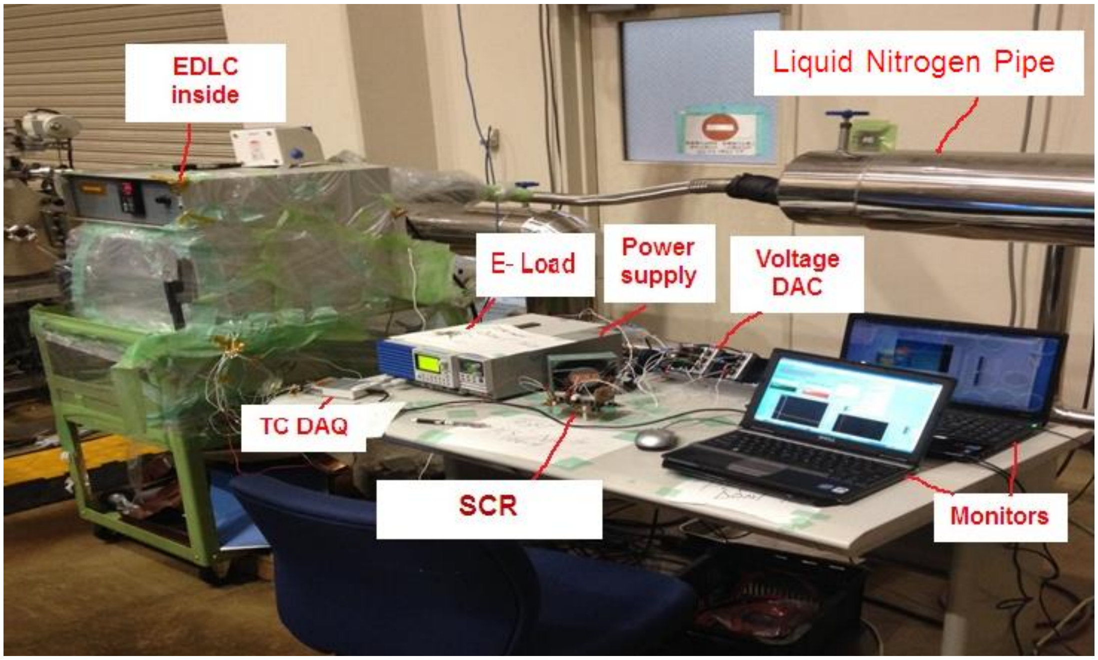

Figure 3:7 Te*st bench for testing at various temperatures in atmosphere.*

Table 3:7 Vacuum chamber specification

Chamber wall material	Stainless still (SUS304)
Size	Length: 100cm, Diameter: 30cm
Ultimate vacuum	1.0 x10-3 Pa ~ 1.0 X 10-5 Pa
Operating temperature range	-150°C ~ +150°C
Thermal Input	Without rail: Cold = Shroud, Hot = Sheet heater With rail: Cold = Shroud, Hot = IR Lamp
Temperature measuring sensor	K-Type Thermocouple（10CH）

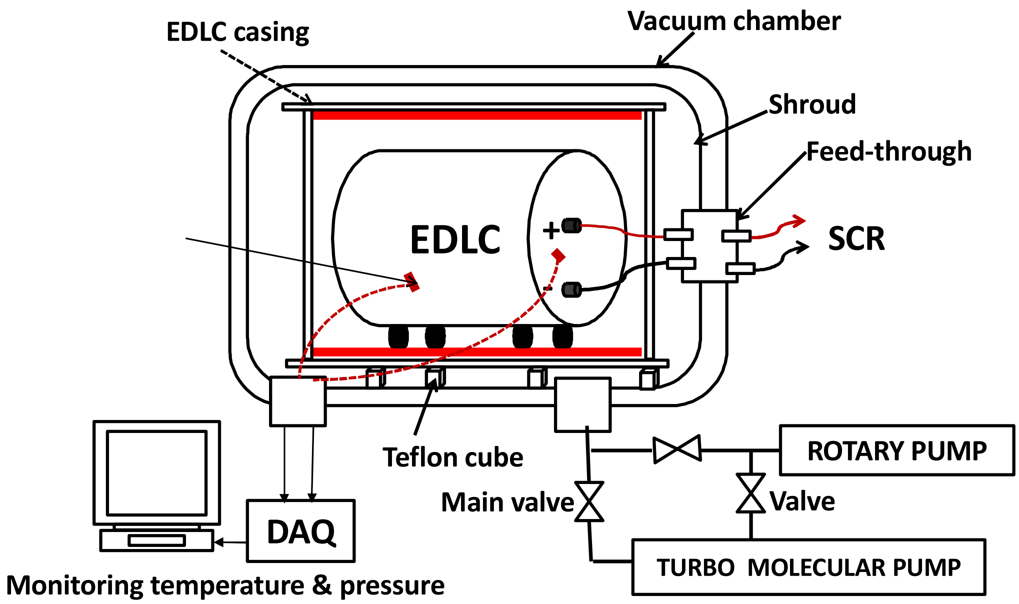

Figure 3:8. *Schematic of the vacuum chamber*

Figure 3.9 shows the schematic for the total ionization dose test on the EDLC cell. This radiation test was conducted using the facilities at the Centre for Accelerator and Beam Applied Science at Kyushu University, Japan. Imposition of cobalt 60 (gamma ray) to the capacitor cell was carried out to verify whether it could withstand a radiation environment. A separation distance of 100 cm was selected between the radiation source and the capacitor to obtain a total ionization dose of 10 krad at the end of 4 h at a dose rate of 2.5 krad/h. This assumes three years total dose of silicon in a simple aluminium sphere of 2mm orbiting in 700km circular orbit of 98 degree inclination (ISO/CD/19683).During the irradiation, the system was operational and underwent charge and discharge operation for 65 minutes and 35 minutes respectively for 1 cycle.

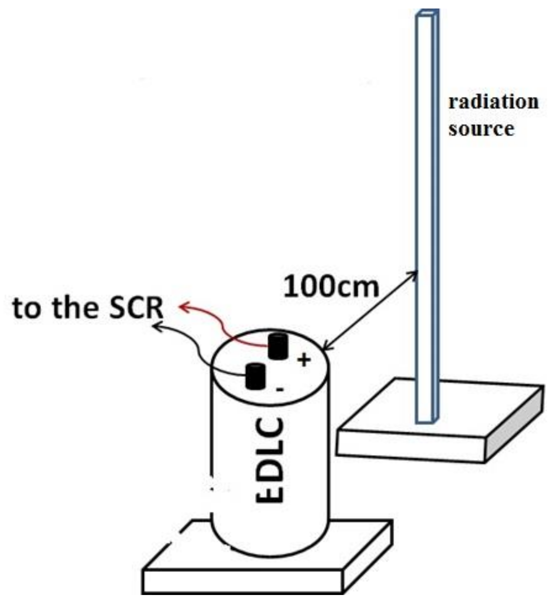

Figure 3:9.*Test schematic for radiation test*

One-week testing was embarked upon to check the durability of the EDLC after undergoing (non-stop) operation for over 168 h (7 days). The test conditions and schematic of the one-week duration experiment are shown in Table 3.8 and Figure 3.10, respectively. Two functional tests of the EDLC in relation to the one-week duration

experiments were conducted: the first, referred to as pre-one-week duration testing, was conducted before the experiment, while the second, here referred to as post- one-week duration testing, was conducted after the seven days experiment. With the performance comparison from the two functional tests, the durability of the EDLC after undergoing one-week duration testing was deduced.

Table 3:8 Test conditions for one-week duration test

Test type	**Power source condition**	**EDLC Status**	**Load condition**	**LabVIEW logic voltage setting (V)**
Functional test	PS supplies power (sunlight simulation)	Charge	Standby	0.5V to 2.4V
	PS cut-off supply (eclipse Simulation)	Discharge	1A/CC mode	2.4V to 0.5V
One-week duration	PS supplies power (sunlight simulation)	Charge	Standby	0.2V to 2.8V
	PS cut-off supply (eclipse Simulation)	Discharge	2A/CC mode	2.8V to 0.2V

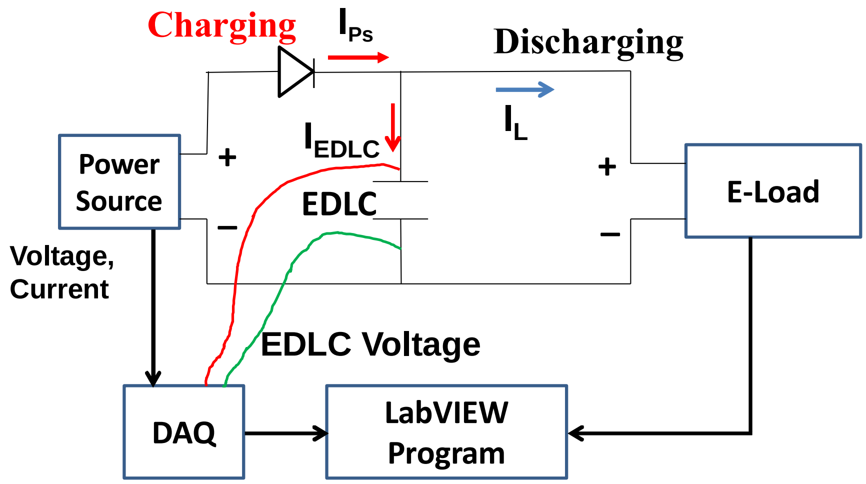

Figure 3:10. *Schematic of one-week duration test*

For the functional tests carried out on or before the one-week duration testing, the PS was 'on' to continuously charge the EDLC under constant current-constant voltage (CC-CV) mode until the EDLC was charged to 2.4 V (the controlling voltage value for the PS to stop supplying power based on the LabVIEW program). The supplying current to the EDLC from the PS is here referred to as I_{ps}, which is the EDLC current (I_{EDLC}) in the charging stage. From this point, the EDLC discharged directly to the load at a constant current of 1 A (at this stage the I_{EDLC} was equal to the load current, I_L, until the voltage dropped to 0.5 V, the value set to trigger the PS 'on' for the charging condition).

For the one-week testing, the PS was ‘on’ to continuously charge the EDLC under CC-CV mode until the EDLC was charged to 2.8 V (the controlling voltage value for the PS to stop supplying power based on the LabVIEW program). From this point, the EDLC discharged directly to the load at a constant current of 2 A until the voltage dropped to 0.2 V (the value set to trigger the power source ‘on’ for the charging condition and trigger ‘standby mode’ for the electronic load).

The specifications of the vibration machine used are given in Table 9. Figure 3.11 shows the sensor positions and vibration direction. The test bench of random vibration is shown in Fig. 3.12. The structural integrity of the EDLC is checked by random vibration testing. Measurement of acceleration was acquired by accelerometers attached to the EDLC. The accelerometer data is acquired by a DAQ system based on the LabVIEW program and was quickly processed to a power spectral density (PSD) pattern after data acquisition. Figure 13 shows the target PSD pattern applied to the jig. **T**his PSD is based on H-2A rocket requirement, though the vibration level (11Grms) was increased to 25 Grms. The base vibration was applied for 110 s in all three directions (X, Y and Z).

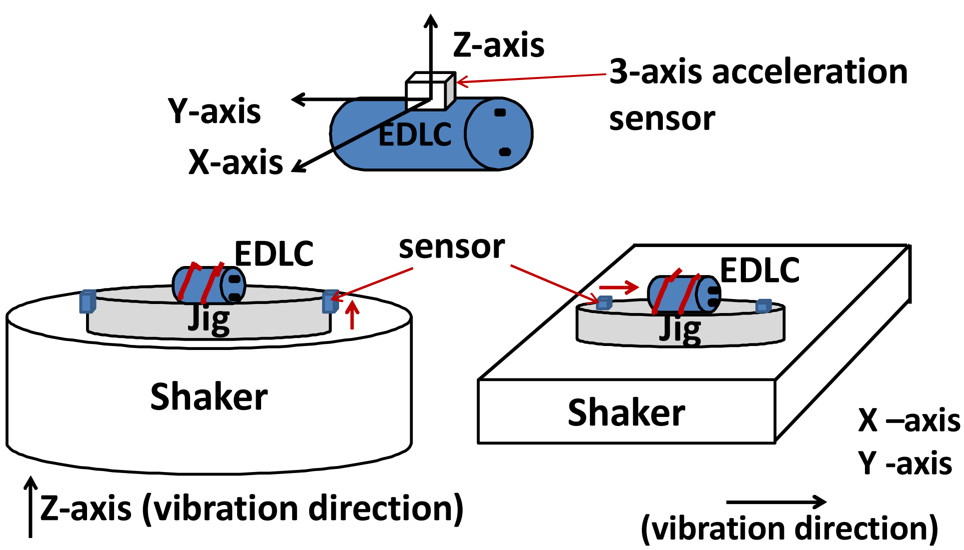

Figure 3:11. *Schematic of one-week duration test. Sensor Positions and vibration directions*

Table 3:9 Specification of vibration machine

<table>
<tr><th>Items</th><th colspan="3">Specifications</th></tr>
<tr><td>Type</td><td colspan="3">F-35000BD/LA36AP (made by EMIC)</td></tr>
<tr><td rowspan="2">Exciting force</td><td>Sine</td><td colspan="2">35.0 kN</td></tr>
<tr><td>Random</td><td colspan="2">28.0 kNrms</td></tr>
<tr><td rowspan="2">No-load maximum acceleration</td><td>Perpendicularity</td><td>Sine</td><td>1060.0 m/s2</td></tr>
<tr><td>Horizontal</td><td>Sine</td><td>460.5 m/s2 (0-p)</td></tr>
<tr><td rowspan="2">Maximum loading mass</td><td>Perpendicularity</td><td colspan="2">400kg</td></tr>
<tr><td>Horizontal</td><td colspan="2">500kg</td></tr>
<tr><td>Horizontal vibration stage</td><td colspan="3">50cm x 50cm</td></tr>
<tr><td rowspan="2">Number of channel</td><td>Measurement</td><td colspan="2">16</td></tr>
<tr><td>Control</td><td colspan="2">8</td></tr>
</table>

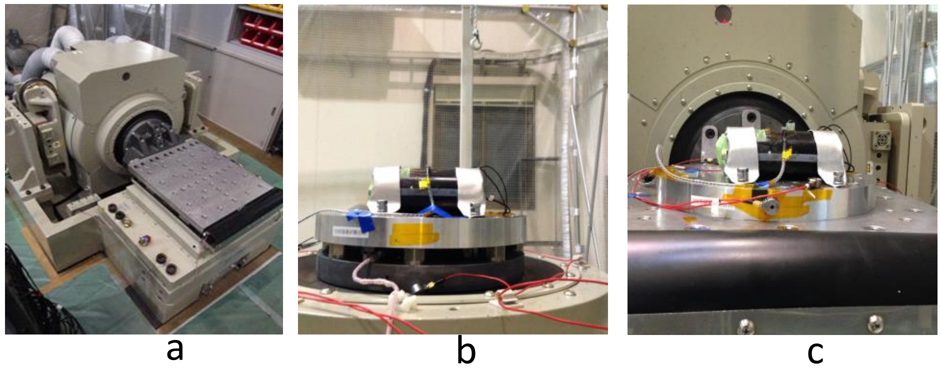

Figure 3:12. *Schematic of one-week duration test. a.) Vibration machine, b.) EDLC on vertical mode of machine, c.) EDLC on horizontal mode of machine*

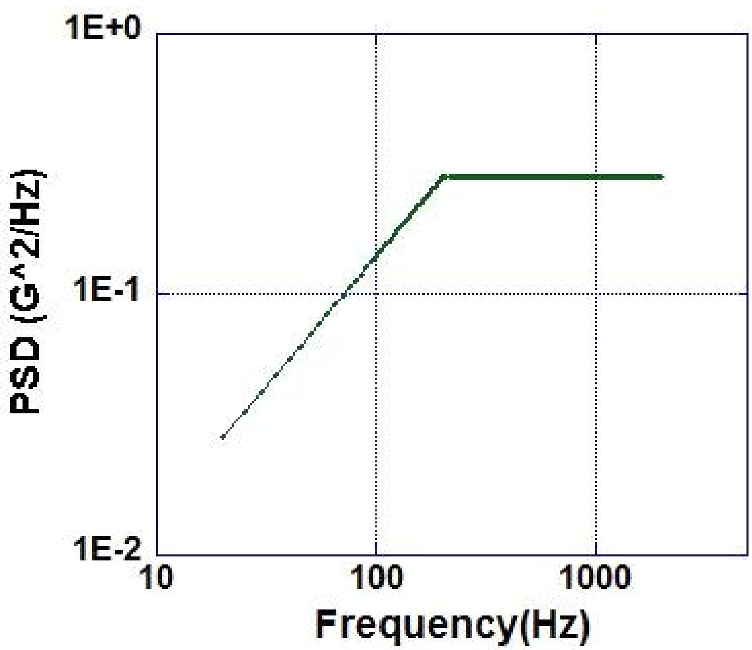

Figure 3:13. *Schematic of one-week duration test . Input profile of the vibration test*

Two functional tests of the EDLC in relation to the vibration experiments were conducted: the first, referred to as pre-vibration testing, was conducted before the experiment, while the second, referred to as post-vibration testing, was conducted after the vibration experiment. With the performance comparison from the two functional tests, the survival of the EDLC after undergoing random vibration was deduced. A three-axis acceleration sensor was firmly glued onto the side of the EDLC, which was tightly screwed on the jig for performance in the vibration test, based on three random tests under the X, Y and Z directions. First, the control accelerometers were placed on the jig top at the edge to perform vibration testing under the axial (Z) direction; after completion of the random test in the Z-direction, the same shaker and support, connected to the horizontal vibration table, were used for the X and Y axes vibration tests (transverse vibration).

In addition to vibration testing, shock testing was also performed. An actual separation shock uses pyrotechnics and its use is limited to only an allowed area. Therefore, the shock testing for nanosatellite components requires a machine free of pyrotechnics. The shock pattern is normally defined by a shock response spectrum (SRS) pattern and the patterns are different from those of the launchers (Masui et al, 2013). The targeted SRS is the requirement of H-2A rocket (satellite separation) that gives 6dB/Oct up to 1500 Hz at 2.5×10^4 m/s^2. A typical launcher requirement was considered. A machine equipped with a hammer (picture shown in Fig. 14) was used. Such a hammer-type shock machine has enough capability for a 1-kg component.

The schematic shown in Fig. 3.14 features a hammer-type shock machine in the configuration. The capacitor was tightly screwed and clipped to the jig using two aluminum bands. At the edge of the jig were two piezoelectric accelerometers positioned for horizontal (0.7303 pC/g) and vertical (0.70245 pC/g) charge output measurement after the hammer impacted on the jig. The voltage output was measured using the DAQ instrument and archived on the PC. The shock was applied twice in two directions, one each for the transversal and longitudinal directions.

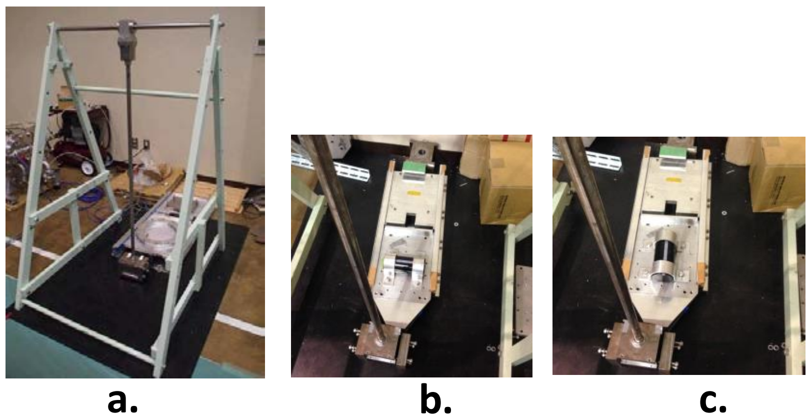

Figure 3:14. *a.)Shock machine, b.) The EDLC transversal positioning, c) longitudinal positioning*

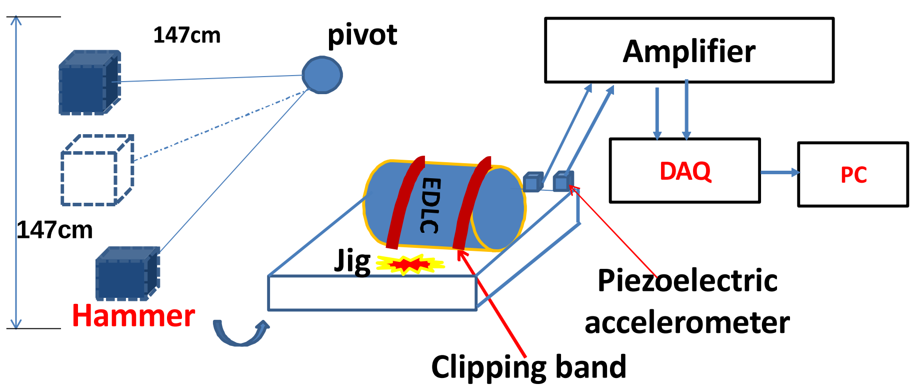

Figure 3:15. *The schematic for shock test*

3.2 Test Results

The graphs in Figures 16 and 17 show three and one cycle(s), respectively, of the EDLC voltage profiles at high temperatures (+65°C). The load scheme was set at a constant power of 1.2 W and an energy of (1.2 W*35*60 s=2520 J) was hence delivered to the load during eclipse periods (35 min). The voltage drop at 3901 s was due to the internal resistance. During the charging time, the voltage was quickly recovered. The charge, Δq (coulomb), is calculated using Equation 3.1, where i and Δt are the delivered current and duration, respectively:

$$\Delta q = \int_0^{\Delta t} i dt \qquad \text{Equation 3:1}$$

The capacitance, C (Farad), can be deduced using Equation 3.2, where q and ΔV are the delivered charge and the difference in voltage, respectively, from the beginning of discharge after the instantaneous drop to the end of the voltage discharge:

$$C = \frac{q}{\Delta V} \qquad \text{Equation 3:2}$$

In this case, we define two types of depths of discharge (DOD), energy and charge, as follows:

$$D.O.D_{energy} = \frac{\frac{1}{2} C \Delta V^2}{\frac{1}{2} C \Delta V_{rated}^2} \qquad \text{Equation 3:3}$$

$$D.O.D_{charge} = \frac{C \Delta V}{C \Delta V_{rated}} \qquad \text{Equation 3:4}$$

where V_{rated} is the rated voltage of the EDLC, i.e., *2.5* V. The relationship between the voltage drop, the actual voltage of the capacitor, V_{EDLC}, and the apparent measured voltage, V_{oc}, is given by Equation 3.5. Note the voltage drop is a product of the internal resistance, ρ, of the EDLC due to an upset of the current flow reversal (from current flow from the power source to the capacitor to current flow from the capacitor to the load as the capacitor starts discharging) and the instantaneous charging current, i_c.

$$V_{OC} = i_C \rho + V_{EDLC} \qquad \text{Equation 3:5}$$

The relationship between the voltage rise, V_{EDLC} and V_{od}, is given by equation 3.6. Note the voltage rise is a product of the internal resistance, ρ, of the EDLC due to an upset of current flow reversal (from current flowing from the capacitor to the load as the capacitor discharges to current flow from the power source to the capacitor as the capacitor starts getting charged) and the instantaneous discharging current, i_d.

$$V_{od} = -i_d \rho + V_{EDLC} \qquad \text{Equation 3:6}$$

$$\rho = \frac{V_{oc} - V_{od}}{I_C - (-I_d)} = \frac{V_{oc} - V_{od}}{I_C + I_d} \qquad \text{Equation 3:7}$$

Applying these equations, we can obtain a capacitance of 3062 (F), $(D.O.D)_{energy}$=20(%), $(D.O.D)_{charge}$=12(%), and an internal resistance of 0.145 Ω. Table 3.10 summarizes the capacitance and the internal resistance of the four test cases listed in Table 4.5. As the temperature decreases, the DOD increases due to an increase in the internal resistance.

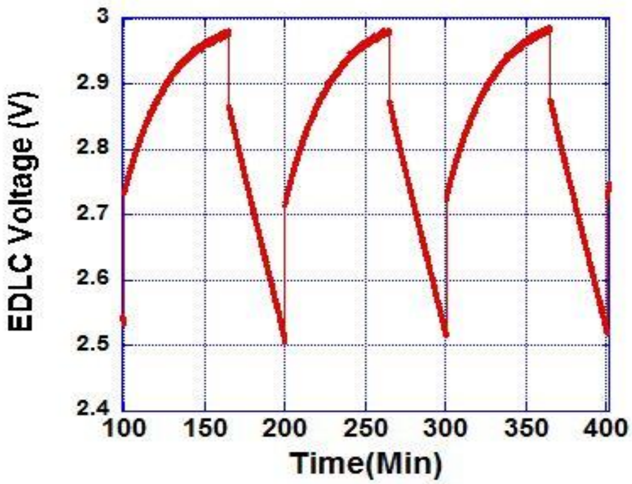

Figure 3:16.*Three cycles of EDLC performance profile at 65°C*

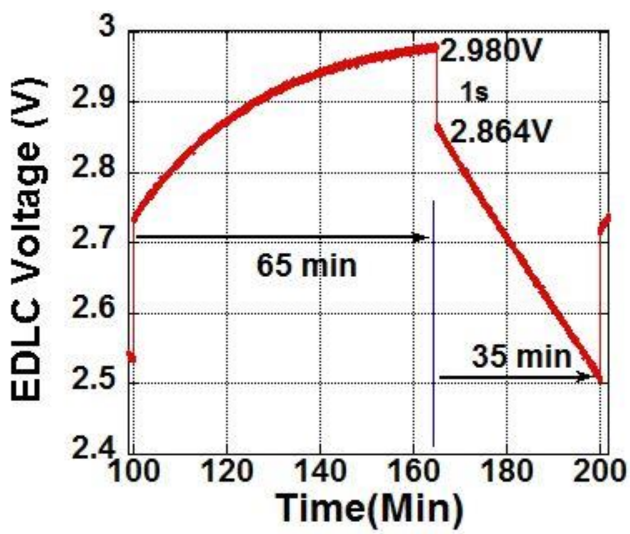

Figure 3:17. *One cycle of EDLC performance profile at 65 degree Celsius*

Table 3:10 . Summary of results

Test Condition	Capacitance (F)	Internal resistance(Ω)
+65°C	3062	0.127
-25°C	3233	0.148
-35°C	3255	0.155
Room temperature in vacuum	3307	0.106
Pre-radiation	3746	0.095
During radiation	3649	0.63
Post radiation(8 days after)	3746	0.094
Post radiation (13 days after)	3746	0.095

From the test results in Table 3.10, above, as there is an increase in internal resistance with a decrease in temperature, so is there a decrease in capacitance with an increase in temperature (comparing the atmosphere variation test +65°C, -25°C and -35°C). Also of note, the capacitance available outside the durability test is within ±10% tolerance from the rated value. In comparing the performance in vacuum with that in atmosphere, no degradation effect was observed due to the vacuum environment. See equation 3.8 for the formula of capacitance available.

$$C_{available} = C_{rated} \pm \frac{\Delta C}{C_{rated}} 100\% \qquad \text{Equation 3:8}$$

where capacitance tolerance, C_t, is given by $C_t = (\Delta C/C_{rated})\ 100\%$ where $\Delta C = C_{available} - C_{rated}$.

3.2.1 Total Ionization Dose on EDLC

Prior to the total ionization dose (TID) test, a preliminary test was conducted to obtain reference data, hereafter referred to as pre-radiation exposure. After the radiation test, series of data were obtained 8 and 13 days after, hereafter referred to as post-radiation exposure. Figure 18 shows a comparison of EDLC performance before ($V_{pre\text{-}rad}$), during ($V_{radiated}$), and after ($V_{post\text{-}rad}$) radiation exposure. During the radiation test, the voltage drop at 3900 s is due to the internal resistance. The internal resistance due to the voltage drop during radiation was 0.63 Ω and is much larger than that in pre- and post-radiation tests (0.095 Ω and 0.094 Ω, respectively). The increased resistance is due to exposure to the radiation. Considering the fact that the dosage rate is much larger than that in LEO by four orders of magnitude, the increased resistance would not occur in orbit. The DOD of the pre-radiation performance is 32%, and 32% was also measured as the post-radiation performance of the cell. From the calculation of the DOD in the four cases, there was no degradation on the capacitor due to radiation exposure.

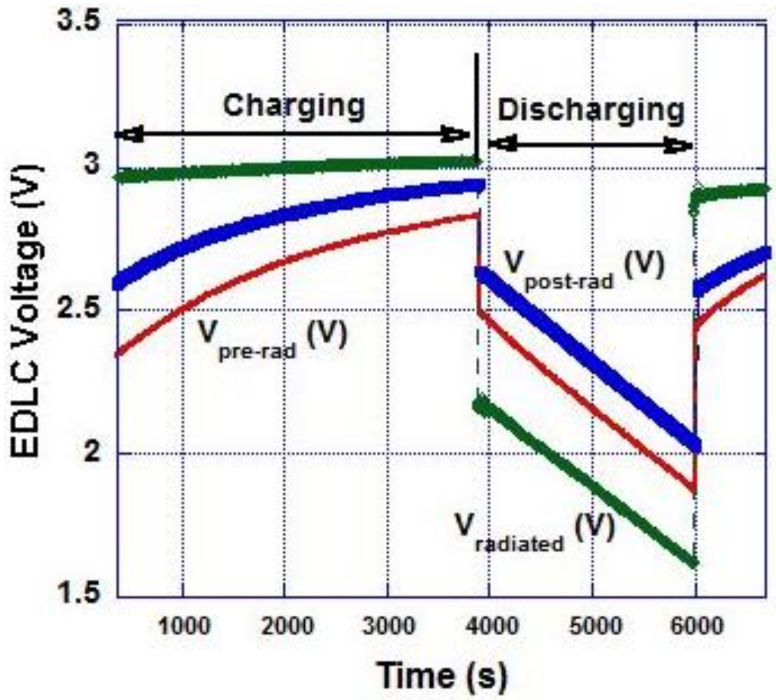

Figure 3:18. *EDLC performance before, during and after radiation exposure*

Table 3.10 summarizes the capacitance and the internal resistance for pre-radiation, during radiation, post-radiation (8 days) and post-radiation (13 days). The EDLC cell was not damaged after exposure to an ionization dose of 10 Krad.

3.2.2 One-week duration test

Figure 19 shows the one-week duration testing results of an EDLC charge/discharge cycle test for seven days. The random spikes seen on the plot were due to errors from the DAQ measurement instrument; this also resulted in little variation in the duration of cycle. The peak values fluctuated within 2.8±0.1 V against the targeted value of 2.8 V due to unknown reason, most likely external noise entering the DAQ.

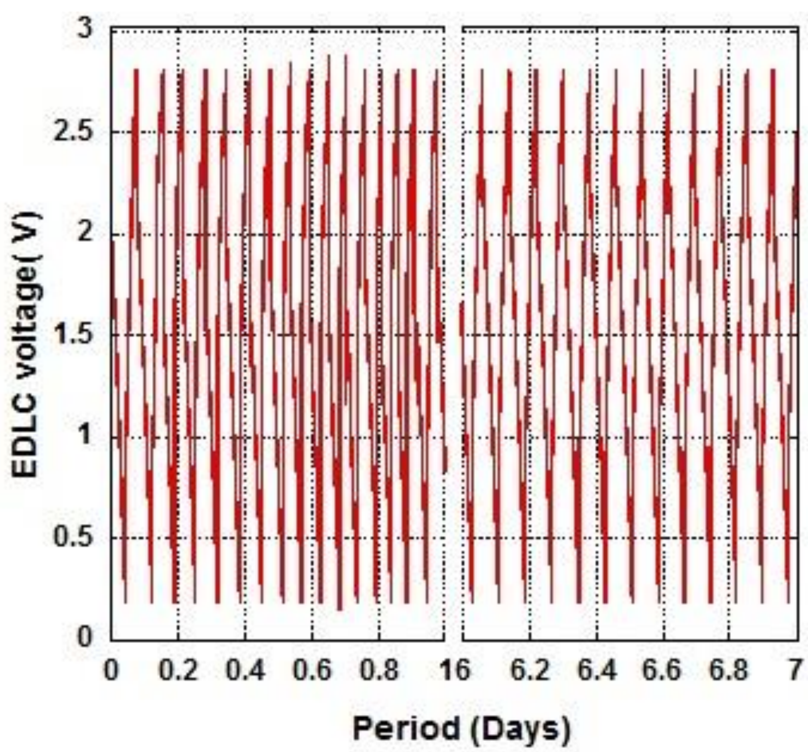

Figure 3:19. *Voltage profile of one-week duration test. The first one day and the last one day profile are shown.*

Figure 20 shows the first charge/discharge cycle of the one-week duration test. The EDLC was charged to 2.82 V within 47 min (0.79 h) and there was a sharp voltage drop within a second to 2.29 V until the capacitor discharged to 0.19 V at a constant current of 2 A within 68 min (1.14 h).

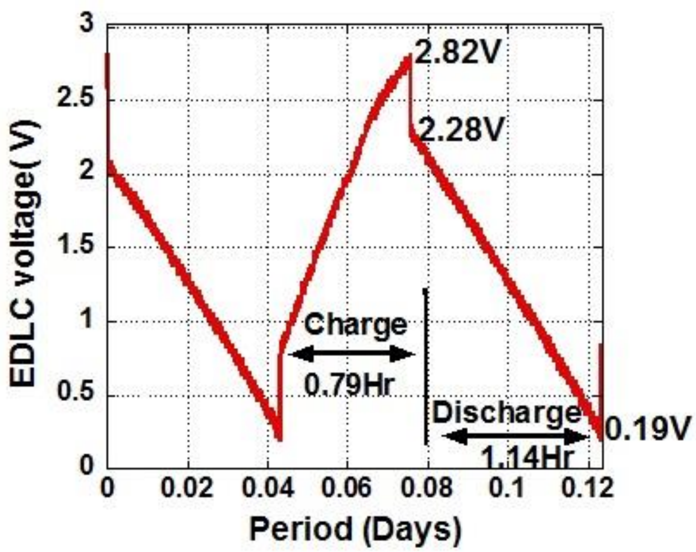

Figure 3:20. *First cycle of the one-week duration*

Figure 21 shows the last cycle on the 7th day of the one-week duration test. The EDLC was charged to 2.81 V within 45.6 min (0.76 h) and there was a sharp voltage drop within 2 s to 2.2 V until the capacitor discharged to 0.18 V at a constant current of 2 A within 65.4 min (1.09 h).

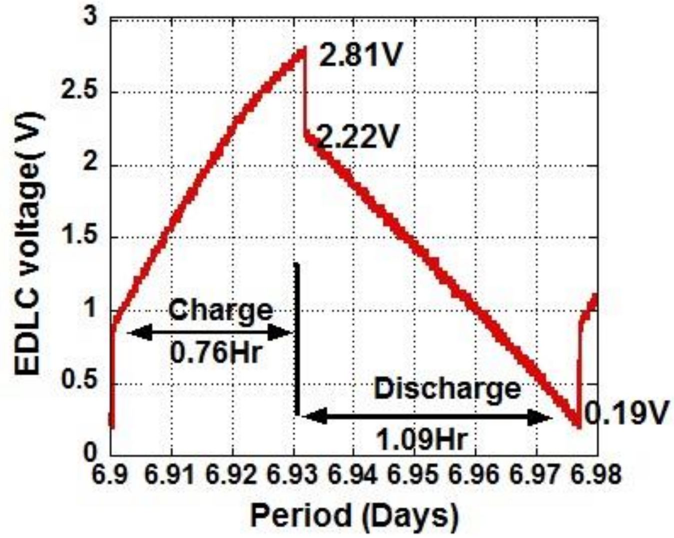

Figure 3:21. *Last cycle on the 7th day of the one-week test*

From the plots, there was almost a consistency in the performance under the same conditions.

3.2.3 Vibration Testing

See Figures 22, 23 and 24 for the response acceleration levels of the EDLC. A summary of the Grms response is shown in Table 3.11. The structural integrity of the cell was not affected even after imposition of Grms value greater than 11G. There was no decline in the performance of the EDLC after post-vibration functional test. Also based on the requirements from launcher's side, the natural frequency in X, Y and Z axis must exceed 100Hz, 50Hz and 50Hz respectively so as not to match with the rocket's natural frequency.

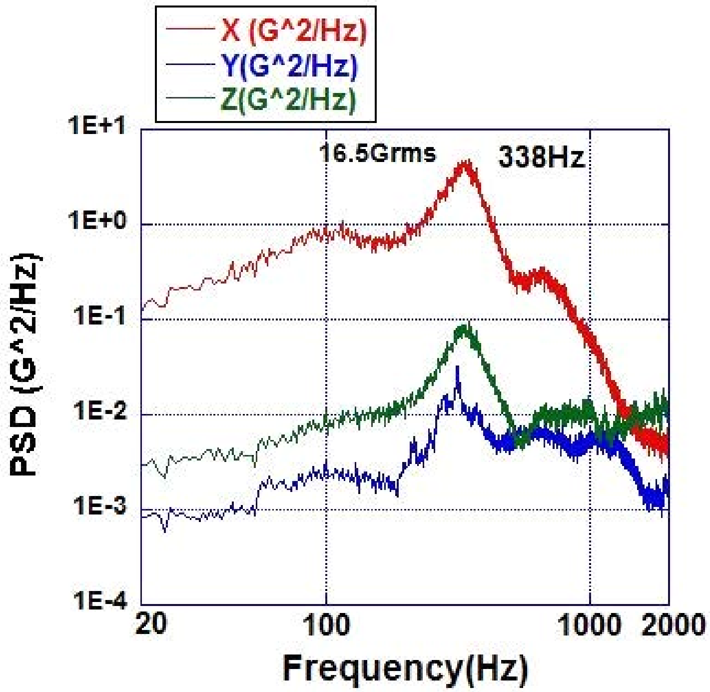

Figure 3:22. *PSD response of EDLC at Random vibration in X direction*

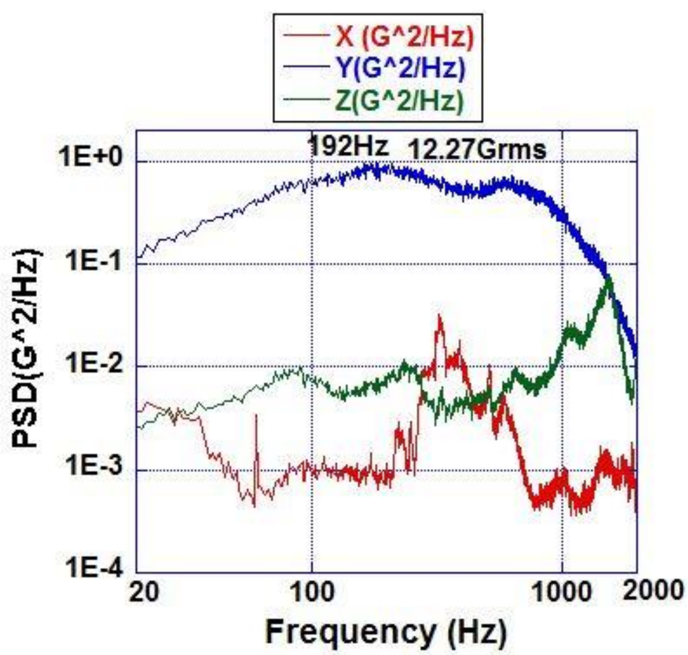

Figure 3:23. *PSD response of EDLC at Random vibration in Y direction*

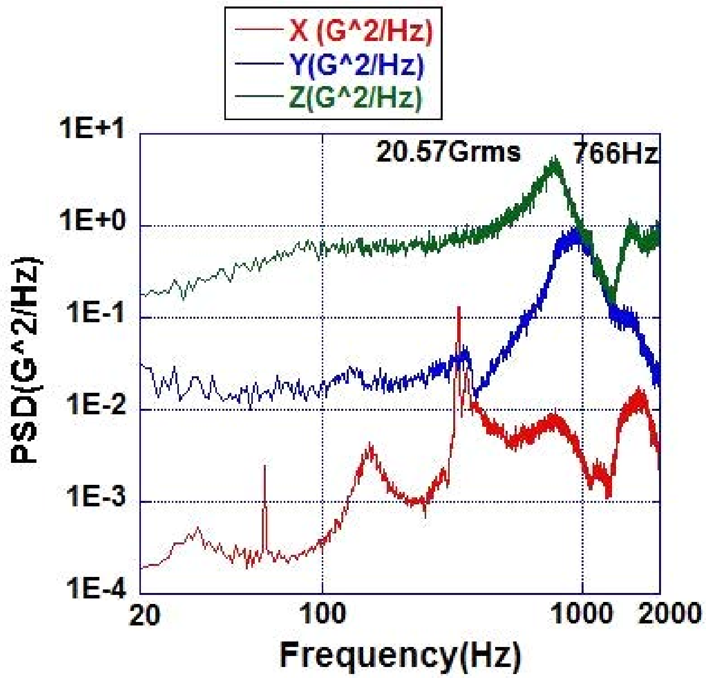

Figure 3:24. *PSD response of EDLC at Random vibration in Z direction*

Table 3:11 : Summary of EDLC Grms response after excitation

Excited axis (25Grms applied)	Grms value of the same axis	First resonant frequency (Hz)
X	16.5	338
Y	12.3	192
Z	20.6	766

3.2.4 Shock test result

Figures 25 and 26 show the shock response spectrum (SRS) values measured at the EDLC for each direction. The directional axis is indicated in Fig. 14(b) and (c). The SRS values at 1,000 Hz and 3,000 Hz were approximately 20,000 m/s^2 and 90,000 m/s^2, respectively. From 100 Hz to 1,500Hz, the SRS follows the requirement of H-2A satellite separation shock test. Beyond 1500Hz, the SRS exceed the rocket requirement, 25km/s^2.

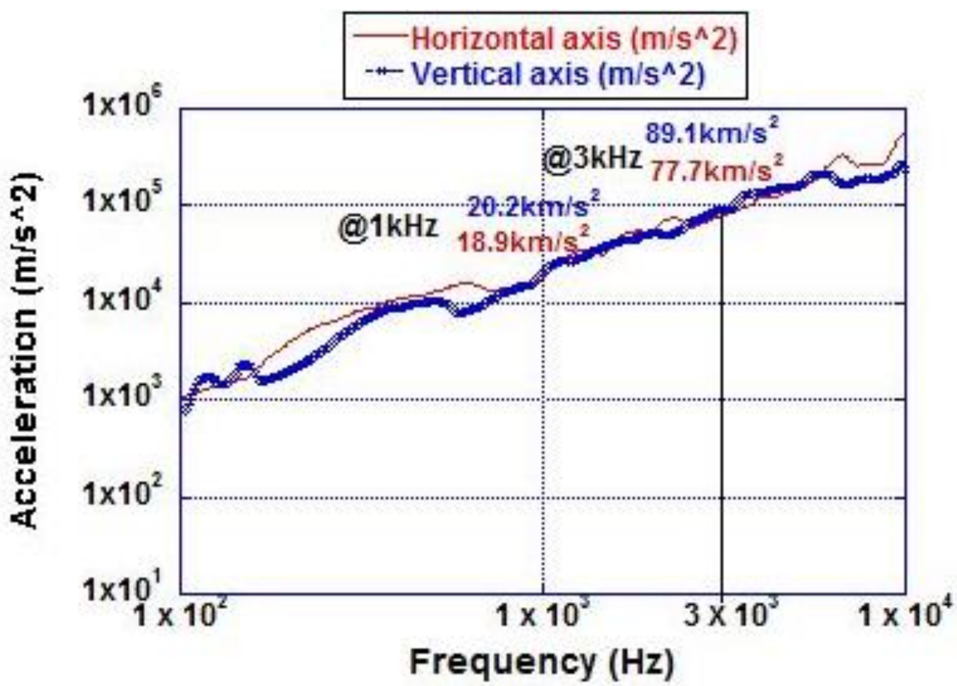

Figure 3:25. *EDLC SRS response on random shock X direction*

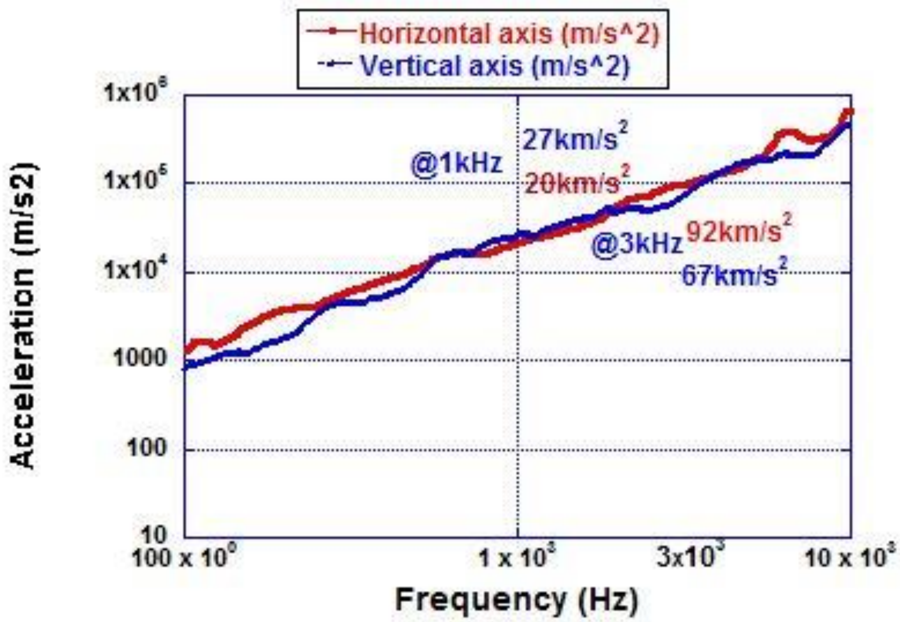

Figure 3:26. . *EDLC SRS response on random shock Y direction*

Table 3.12 summarizes the capacitance and internal resistance of the EDLC cell before and after each durability test. The change of capacitance after durability test was 5%. The change of internal resistance after the entire durability test was 9.7%. These numbers satisfy the experimental criteria of 10% and 15% for the change of capacitance and internal resistance, respectively.

Table 3:12 . Summary of EDLC after durability test

Condition	Capacitance (F)	Internal resistance
As received from manufacturer. (Before the series of environment test)	4183F	0.093
After all the series of environment tests	3974	0.102

3.3 Reliability of the EDLC-based power system

Reliability of EPS is primarily governed by the following factors:

1. Orbital illumination, illumination and temperature of solar cells, whether body mounted or solar cells. This translates to the generating capacity of the power system.
2. Performance deterioration of solar cells due to space environmental effects.
3. Performance deterioration of the energy storage during or after undergoing cycling.
4. Charge/discharge management circuit
5. Redundancy complexity

Since the performance of the solar cell is not directly controlled or related to the storage component. The examining factors that can be used in comparing EDLC -based system and battery-based EPS are the performance of the storage components after cycling, charge/ discharge management and complexity of redundancy.

Since EDLC has more multicycle capabilities than traditional batteries [see table 3.1], the chances of failure is lesser. The EDLC –based system unlike the traditional battery based system has simple charge-discharge management since the charging current of EDLC is unlimited. The EDLC based power system can simply be parallel connection for a backing up system unlike the traditional battery based system which is complex since the mitigations in the series-parallel connections have to be duplicated for each redundancy.

On a general note, reliability depends on the context of design. In nanosatellite development especially those concerning school based projects there is less emphasis on reliability since profitability is questionable.

Ambitious projects for commercial missions can accommodate the robustness, wide operating temperature and other attractive attributes of EDLC to enhance reliability of spacecraft power systems now and the nearest future.

3.4 System comparison of EDLCs and traditional batteries

The table 3:13 below shows the system comparison of EDLCs with other nickel and lithium based batteries.

Table 3:13 EPS comparison with EDLCs and traditional batteries

EPS type	Ni-Cd based	NiMH based	Lithium-ion based	EDLC based
Operating temperature [oC]	0 ~+45	0 ~ +45	-20 ~ +60	-40 ~ +65
Charge /discharge management	Moderate	Moderate	complex	Simple
Leakage mitigation	Necessary	Necessary	Necessary	Necessary
EPS mass percentage in a system(%)	20	20	20	35
Reliability for 1-2 years (%)	70-80	70-80	80-90	>90
Reliability for >2 years (%)	<70	<70	<70	>90

3.5 Conclusions

Electric double-layer capacitors (EDLCs) are promising as robust and easy-to-handle power storage components for nanosatellites. The disadvantage of small gravimetric and volumetric energy densities is constantly diminishing. It is only a matter of time until the energy densities become comparable to those of Ni-MH batteries. It is worthwhile to investigate the strength of EDLCs against the space environment. A series of environmental tests was carried out on a COTS EDLC. The results of high-temperature

(+65°C), room-temperature, low-temperature (-25°C, -35°C) and vacuum tests showed no serious performance degradation of the EDLC under those conditions. Furthermore, total dose ionization, one-week duration, vibration and shock tests were conducted. The results from the EDLC post-functional performance show the cell survived the test, which implies that it is robust. The overall conclusion is that EDLCs can be used on nanosatellites without any thermal and mechanical protection. Also, based on the development trend of the energy density of COTS capacitors, it is projected that, in or before the year 2018, COTS EDLCs with a volumetric density as high as 12 Wh/l will flood the market and, expectedly by 2024, EDLCs with a gravimetric density as high as 12 Wh/kg will be on the market too. This will be revolutionary, especially for LEO missions or nanosatellites. If we accept a volume increase of 50% compared to Ni-MH batteries, EDLCs can even be used now as power storage units for nanosatellites.

In the present study, the longest cycle was one week, far shorter than the typical mission duration of nanosatellites, six months at shortest. The long-life charging and discharging test is a task of future work as well as using a more realistic power profile. Also there is a future plan to perform temperature cycle test. As the present tests were done only on a particular type of EDLC, in the very near future we will test other types of state-of-the-art EDLCs on the market.

3.6 References

Alkali, M., Edries, M.Y., Khan, A.R., Masui, H., Cho, M., "Design Considerations and Ground Testing of Electrical Double Layer Capacitor as an Energy Storage Component for nanosatellite", Journal of Small Satellite (JoSS)[published in October issues, Vol. 04, No. 02 (October 2015) pp. 387-405].

Alkali,M.,Edries,M.Y.,Khan,A.R.,Masui,H.,Cho,M(2015)Environment Test Campaign of Commercial-off-the shelf Components of Electrical Double Layer Capacitor for space Use, present at the 34th ISAS/JAXA Space Power Symposium ,6th March,2015, Institute of Space and Astronautical Science (ISAS)/Japan Aerospace Exploration Agency (JAXA) , Sagamihara, Japan

Alkali,M.,Edries,M.Y.,Khan,A.R.,Masui,H.,Cho,M(2014):Preliminary Study of Electric Double Layer Capacitor as an Energy Storage of Simple Nanosatellite Power System, presented at the 65th International Astronautic Congress (IAC), 29th September to 4th October 2014, Toronto, Canada.

Alkali,M.,Edries,M.Y.,Khan,A.R.,Masui,H.,Cho,M(2014):Performance evaluation of electric double layer capacitor as energy storage component of micro/nanosatellite on the imposition of varied temperature and vacuum conditions, presented at the 50th

AIAA/ASME/SAE/ASEE Joint Propulsion conference 28-30 July 2014, Cleveland Convention Centre, Cleveland ,Ohio.

Hyder,A.K, (2003): A century of Aerospace Electrical Power Technology, Journal of Propulsion and Power, Vol. 19, No. 6 (2003), pp. 1155-1179.

Patel, M.R (2005): Spacecraft Power Systems, CRC Press, ISBN 0-8493-2786-5.

Ratnakumar, B.V, Smart, M.C, Kindler, A, Frank, H, Ewell, R, Surampudi,S(2003): Rover, Journal *of Power Sources*, *Volumes 119–121*, *1 June 2003*, *Pages 906-910*

Masui,H., Hatamura,T.,Cho,M. (2013): Testing of Micro/Nano Satellites and Their On-orbit Performance

Presented at the 27th Annual AIAA/USU conference on small satellite, August 10 – 15, 2013, Utah State University, Logan, UT, USA

ISO/CD/19683: Space systems —Design Qualification and Acceptance Tests of Lean Satellites and Units

Alkali. Gana,C., Idrisu,D.,Cho,M.(2014), SPIAS: Evolutionary approach for Space Science Education Development in Africa,IOSR Journal of Research & Method in Education (IOSR-JRME) e-ISSN: 2320–7388, p-ISSN: 2320-737X Volume 4, Issue 1 Ver. II, PP. 22-27

Alkali,M., Motohata,Tatsuo,S,Masui,H.,Cho,M. (2013):Supercapacitor: Testing it's practicability as power storage unit of a nanosatellite presented at the 5th Nanosatellite symposium 20-22, November 2013, Tokyo

Buchen, M., "2014 Nano/Microsatellite Market Assessment," SpaceWorks Enterprises, Inc. (SEI) Atlanta, GA (2014), http://www.sei.aero/eng/papers/uploads/archive/SpaceWorks_Nano_Microsatellite_Market_Assessment_January_2014.pdf

Edries, M.Y, Alkali. and Cho,M(2014) :Design and evaluation of a Nano-satellite Battery Charge Regulator (BCR) based on a simple Maximum Power Point (MPP) tracking control, presented at the 58th Japan Society of Aeronautical and Space Sciences (JSASS) conference, 12-14 November 2014, Nagasaki, Japan

H. Heidt, et al., (2000): CubeSat: A New generation of picosatellite for education and industry low-cost space experimentation, in: Proceedings of the 14th Annual/USU Conference on Small Satellites, Logan, USA, 2000.

Nakasuka, S., Kawashima, R., (2012): Micro/Nanosatellite Activities by Japanese Universities and Vision towards International Contribution, at the Committee on the Peaceful Uses of Outer Space: 55th session 6-15 June 2012.

CHAPTER FOUR: LABORATORY VERIFICATION OF EDLC BASED POWER SYSTEM FOR A CUBESAT MISSION

4.1 Introduction

Development and launch of satellites for space research or application based missions are very expensive ventures for research institutions and outside the budget of universities. With the emergence of CubeSats sixteen years ago, the two founders, Jordi Puig-Suari and Robert Twigg, have demonstrated that low cost and design minimization could be achieved for satellites by using a standardized deployer compatible with launchers, which is known as the Poly Picosatellite Orbital Deployer (P-POD)[Alkali,2015;Nanosat,2014].

Based on the standard, the basic CubeSat size is 10cm×10cm×10cm and it is referred as 1U CubeSat. By extension, a 2U CubeSat as a size of 20cm×10cm×10cm. 1U and 2U CubeSat's masses are defined to be maximum 1330g and 2600g respectively [Alkali,2015;JAXA website]. At the beginning, many universities throughout the world took and are still utilizing the opportunities to develop and launch their CubeSats. The CubeSats, just like other satellites, require electrical power system (EPS). In order to efficiently utilize these satellites' missions, the EPS must be efficient, durable, and simple. The EPS must be designed to meet the power requirement of the missions, while considering its robustness to the harsh space environment. A typical CubeSat must consist of an EPS that has the capability to generate, store, and distribute electrical energies to the entire bus and payloads [Nakasuka,2012;Alkali,2013].

The power system unit includes solar cells, Power Control Unit (PCU), and energy storage unit. The solar cells generate electricity from the received solar illumination, the

PCU are designed for power regulation and distribution, and the energy storage unit commonly uses batteries for providing power complement in critical situations and/or during eclipse.

Energy storage is a critical unit of a CubeSat's power systems, which currently uses primary or secondary batteries. Electric Double Layer Capacitors have extremely high capacitive densities compared to the ordinary capacitors hence have traits that are very attractive for applications that are hitherto reserved for batteries. Unlike batteries, EDLCs are capable of directly storing energy without conversion from chemical to electrical energy. Another attractive quality is that EDLCs are safe to handle since there is no risk of explosion, whereas explosion is a critical concern with chemical batteries on which temperature effects need to be carefully considered. EDLCs can quickly be charged and discharged and due to the absence of chemical processing, they can undergo several thousands of cycles (charge and discharge) with minimal degradation [Alkali, 2014; Burke 2000].

Batteries or any other energy storage components are primarily characterized by their energy densities, operating temperature, power density, and number of cycles capability. The commonly used batteries are nickel-based batteries such as NiCd and NiMH [Conway, 1999; Sato; Shimizu, 2009; Fetchera, 2010]. Table 4.1 compares NiCd, NiMH, and EDLC energy storage performances.

The robustness and simplicity is very attractive as a power storage device of universities or school educational CubeSats that do not require the cutting edge mass and volume saving technology to do an advanced mission. The educational satellite puts emphasis on learning an entire process of satellite system life cycle from conceptual design to disposal. For such a mission, only the basic functionalities such as beacon, telemetry and perhaps earth image capture are necessary, requiring less energy. Having a satellite working in space is rather more important than doing an advanced mission. Therefore, for such CubeSats, EDLC may be an attractive solution as the power storage device.

The purpose of the present paper is to simulate and verify the performance of the state of the art (SoA) EDLC as an energy storage system of satellite considering a simple CubeSat mission.

Table 4:1. Comparison of NiCd, NiMH, and EDLC performances

Storage Type	**NiCd**	**NiMH**	**EDLC**
Rated voltage [V]	1.2	1.2	2.85
Gravimetric energy density [Wh/kg]	60	120	7.34
Volumetric energy density [Wh/l]	~60	~150	~8
Gravimetric power density [W/kg]	150	500	14000
Operating Depth of Discharge [%]	20	20	~100
Operating temperature [°C]	0 ~ +45	0~ +45	-40 ~ +65

4.2 EDLC based Power designated cubesat

Specifications of EDLC based EPS designed for CubeSat are shown in Table 4.2. The CubeSat shall be compatible with J-SSOD (Japanese experimental module Small Satellite Orbital Deployer). J-SSOD is a mechanism for the deployment of small satellite based on the CubeSat design specifications that serves as means of transferring satellites from the Japanese experiment module Kibo’s airlock onboard ISS to space environment and then onward for release to the orbit[Alkali,2015].

CubeSats release from the International Space Station (ISS), in comparison to satellites launched from launcher, is performed in less severe conditions since they are launched as a part of pressurized cargo to the ISS and the launch environment, such as vibrations level, is minimal. Release from the ISS also increases the number of launch opportunities for CubeSats since various vehicles, can dock to the ISS. Moreover, the ISS crews can perform a checkout of the CubeSat prior to its release and hence higher reliability can be achieved [Alkali, 2015].

Table 4:2. A typical mission Specifications

Size	1U
Mission	Technical demonstration of SoA EDLC as power delivery.
ISS Orbit altitude	388km
Inclination	52^{o}
Orbit period	92min
Sunlight period	56 min
Eclipse period	36 min
Mission duration	250 days
Required power during eclipse	0.62W
Number of charge–discharge cycles	3887

We assumed that the satellite has a passive attitude control via a permanent magnet only. The axis of the magnet is normal to one face without solar cells (-X face). The permanent magnet axis is aligned with geomagnetic field. Using a hysteresis damper, the libration motion is suppressed. The satellite spins around the permanent magnet (i.e. geomagnetic field) axis. This attitude control scheme is the simplest one adopted by many university satellites. From this assumption, illumination to five solar cell mounted panels was calculated. Here in a cycle, the first 56 minutes based on the illumination which was responsible for power generation and at the last 36 minutes there was cut-off due to eclipse or sun outage, as shown in the Figure 4.1.The power budget requirement for the EDLC based designated CubeSat is considered as 0.62W. The details are shown in Table 4.3. At an altitude of 388km, illustrated in figure 4.1, the satellite's orbit period, eclipse and sunlight time in orbit cycle was calculated.

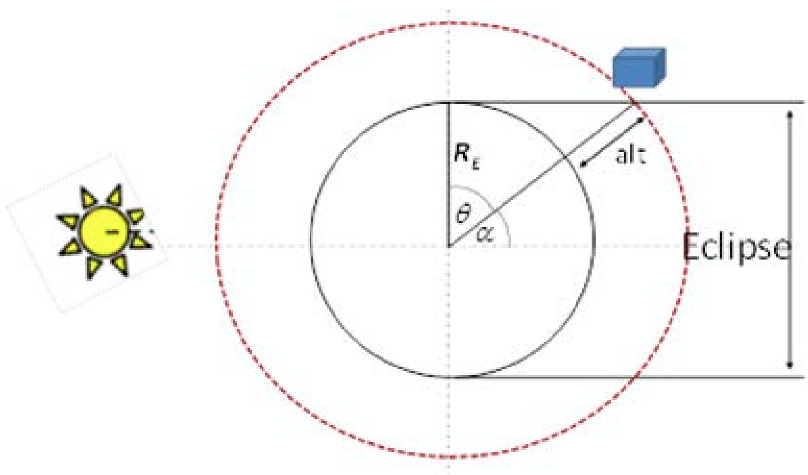

Figure 4:1 Satellite orbit cycle [Alkali, 2015]

Semi major axis, a is given by,

$$a = R_e + \text{alt.} = (6,378.16 + 388)km = 6,766km \qquad \text{Equation 4:1}$$

Where Re, is the radius of the earth (6,378.16km) and alt. is the selected altitude (388km)

The mean motion, n is deduced as shown in equation 4:2

$$n = \sqrt{\frac{\mu}{a^3}} = 0.0011343 rad / s \qquad \text{Equation 4:2}$$

Where μ ($398600.5 km^3/s^2$) is the universal constant,

1 orbit (2π rad)

Orbit time is thus calculated as shown in equation 4:3

Orbit time =2π rad/0.0011343rad/s=5539 sec = 92 minutes Equation 4:3

$$\theta = \cos^{-1}\left(\frac{R_e}{R_e + alt}\right) = 0.34\text{rad}$$

$$\alpha = \left(\frac{11}{2} - 0.34\right)\text{rad} = 1.23 \text{ rad}$$

Eclipse time is calculated as shown in equation 4:4,

$$t_{eclipse} = \frac{2\alpha}{n} = \frac{2.461 rad}{0.0011343 rad/s} = 36 \min utes \qquad \text{Equation 4:4}$$

Hence sunlight time is calculated as shown equation 4:5,

$$t_{eclipse} = (92 - t_{eclipse}) \text{mins} = 56 \min s \qquad \text{Equation 4:5}$$

Table 4:3. Power Budget of the CubeSat

Sub-system	**Operation Mode**	**1 orbit time (Minute)**	**Duty cycle**	**Peak Power (W)**	**Average Power (W)**
OBC	Nominal	92	100%	0.4	0.4
Camera	ON	5	5.4%	0.3	0.0162
	Standby	87	95%	0.02	0.019
COM		10	10.8%	0.8	0.0864
Beacon		92	100%	0.1	0.1
Total				**1.62**	**0.62**

The average power, P_{av} in Table III were calculated using equation 4:6

$$P_{av} = P_{peak} \times DT$$ Equation 4:6

Where P_{peak} is peak power and DT is duty cycle given by equation 4:7

$$DT = \frac{Operating\_time(\text{minutes})}{orbit\_period(\text{minutes})} \times 100\%$$ Equation 4:7

The duty cycle is operating time in the orbit period of 92 minutes. The camera operates to capture images for 5 minutes only. The considered 1U CubeSat has six panels with dimensions of 10cm×10cm .One of the panels is dedicated for the mission (for example, camera). Rests of the five panels are for solar cells in 2P combination (2 solar cells connected in parallel). The bus solar cells are triple junction solar cells and they are body-mounted to the CubeSat as shown in Figure 4.2. Each solar cell's average open circuit voltage is 2.70V with a maximum output voltage of 2.41V for an output current of 0.504A. Table 4.4. Shows the solar cell specifications at the beginning of life (BOL). A parallel combination of 2 cells gives a maximum output voltage of 2.41V with a current of 1.008A.

Table 4:4. Specifications of a triple junction solar cell

TYPE: TJ SOLAR CELL 3G30C - ADVANCED	
Typical Electrical parameters	
	BOL
Average Open Circuit, Voc [V]	2.7
Average Short Circuit, Isc [A]	0.52
Voltage at max. Power, Vmp [V]	2.41
Current at max. Power, Imp [A]	0.5
Average Efficiency ŋbare (1367 W/m2) [%]	29.5
Average Efficiency ŋbare (1353 W/m2) [%]	29.8

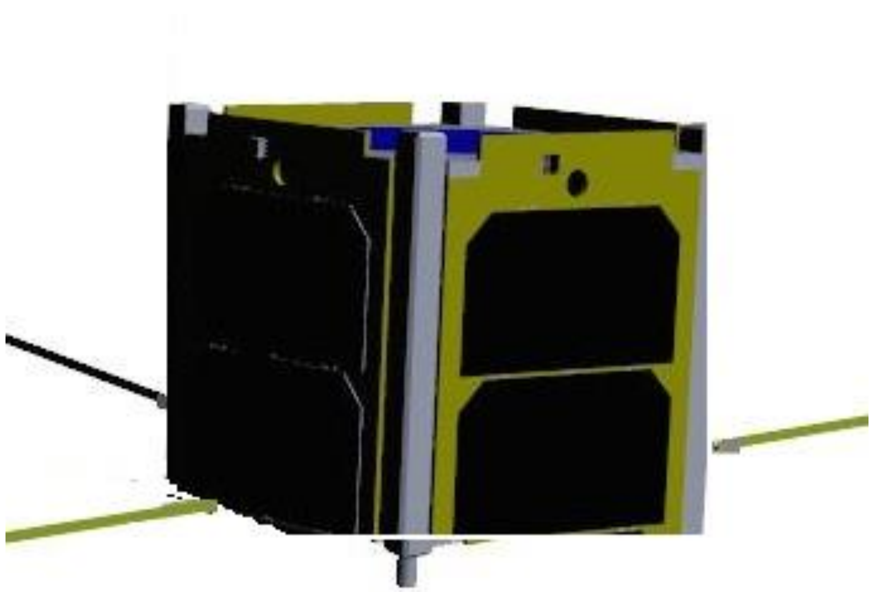

Figure 4:2 Body mounted solar cells on 1u cubesat [Alkali, 2015]

The topology used in this analysis is direct energy transfer (DET), which it is simple, efficient and reliable and the bus voltage is dependent on the state of charge of the EDLC [Vallado,2001].The loss is recorded only at the diode and conversion points only.

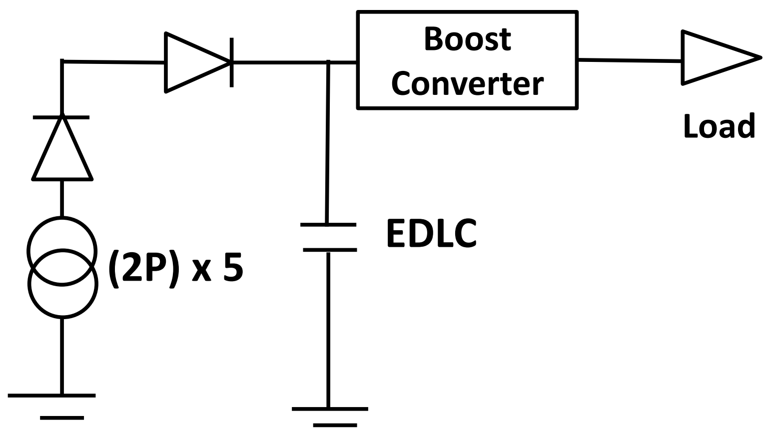

Figure 4:3 The simplest EPS topology [Alkali, 2015]

The power system is simplified with commercial off the shelf components (COTS) that are driven with intent of achieving low cost and quick delivery. The system is configured with cheap COTS diode (using lessened components and simplicity) that serves as the entity for the charge regulator and as protection to prevent backflow of current from the EDLC to the power generator instead of the load.

Commercially available Maxwell EDLC cell (Part No. BCAP1200P270) with a voltage of 2.7V and Capacitance of 1200F was selected. The photograph and specification is shown and given in Figure 4.4 and Table 4.5, respectively.

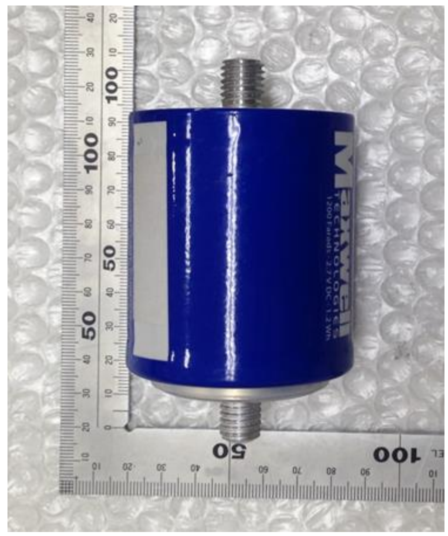

Figure 4:4 Photograph of EDLC used for LEO orbit simulation [Alkali, 2015]

Table 4:5. Specification of the EDLC cell

Part No.	**BCAP1200 P270**
Rated Voltage [V]	2.7
Capacitance [F]	1200
Operating temperature range [°C]	-40 to +65
Size	7cm (L) x 5.7cm (ϕ)
Mass [kg] (with two end bolts)	0.294

Series of tests were performed on the same series of EDLC with a different capacitance of 3400F (Part no. BCAP3400P285) to confirm the durability. The tests were thermal vacuum, random vibration, thermal cycling and gamma ray radiation. EDLC survived all the tests confirmed by the pre- and post-charge/discharge pattern. Details were explained in chapter 3.

Power delivery to the load during eclipse time of 36 minutes is 0.62W. The expected energy in eclipse time by the EDLC is thus;

$$EDLC_{energy} = 0.62W \times (36 \times 60)s = 1.34kJ \qquad \text{Equation 4:8}$$

The potential energy of the capacitor at rated voltage of 2.7V, E_{rated} is thus calculated by equation 4:9,

$$E_{rated} = 0.5C_{rated}V_{rated}^2 = 4.4kJ \qquad \text{Equation 4:9}$$

Since it is expected that the maximum accruable voltage of the solar cell is 2.41V with a drop 0.2V due to the diode, hence, the maximum chargeable voltage level or $E_{rated\_new\ of}$ the EDLC shall be 2.21V. The attainable energy of the cell under this condition shall be

$$E_{rated\_new} = 0.5C_{rated}V^2_{rated\_new} = 2.9kJ$$ Equation 4:10

Discharging 1.34kJ from realizable energy of 2.9kJ, the energy at the end of discharge at 36 minutes is 1.56kJ.

$$\Delta E = 2.9\text{kJ} - 1.34kJ = 1.56kJ = 1560J$$ Equation 4:11

$$V_{eod} = \sqrt{\frac{1560}{0.5C}} = 1.61V$$ Equation 4:12

With this voltage we were able to set the converters operating voltage level.

4.3 Test set up and results

Two cases (referred as Case-1 and Case-2) were simulated for emulation the EDLC based cubesat power system. For Case-1, to emulate two solar cells in parallel connection as seen in the design, the two solar array simulators (SAS-1 and SAS-2) were connected in parallel; hence a common voltage profile and combined current, I_{SAS} duly obtained from (I_{SAS-1} & I_{SAS-2}) were supplied to the single load as shown in Figure 4.5. For Case-2, two electronic loads of 3.3V and 5V buses were used as shown in Figure 4.6. Two boost converters (Part No. LM2623MM, TEXAS INSTRUMENT) were used for the 5V and 3V bus.

For the emulation test the equipment and set up two Agilent's Solar Array Simulator (SAS) [E4351B; 4A, 480W] were used to simulate the solar power profile. The two SAS and EDLC were interfaced to one or two load (PLZ164WA, KIKUSUI) for Case-1 & Case-2 respectively and the EDLC through the diode. Simulation of the charge/discharge cycles was implemented using LabVIEW program, which controlled the operational mode of the electronic load and the SAS (ON/OFF). DAQs operated by LabVIEW program were used to control charge/discharge cycles (whether time or voltage controlled) and recorded the voltage and current profile of both SASs and loads. The charge (56 minutes) and discharge time (36 minutes) conditions were time controlled [Alkali, 2015].

Table 4.6. shows the functionality test conditions for LEO orbit.

Table 4:6. LEO orbit emulation Charging/Discharge test Conditions

Condition	**Period [minutes]**
SAS supplies power (ON mode) (Sunlight simulation)	56
SAS power cut-off (OFF mode) (Eclipse simulation)	36
One cycle	92
1 Day (16 cycles)	1472

For data acquisition, a test was performed to emulate the condition when the CubeSat is released from ISS at an orbit altitude of 388km with an orbital period of 92 minutes (56 minutes in sunlight time and 36 minutes during eclipse). A simulation of sixteen (16) cycles was conducted for a day.

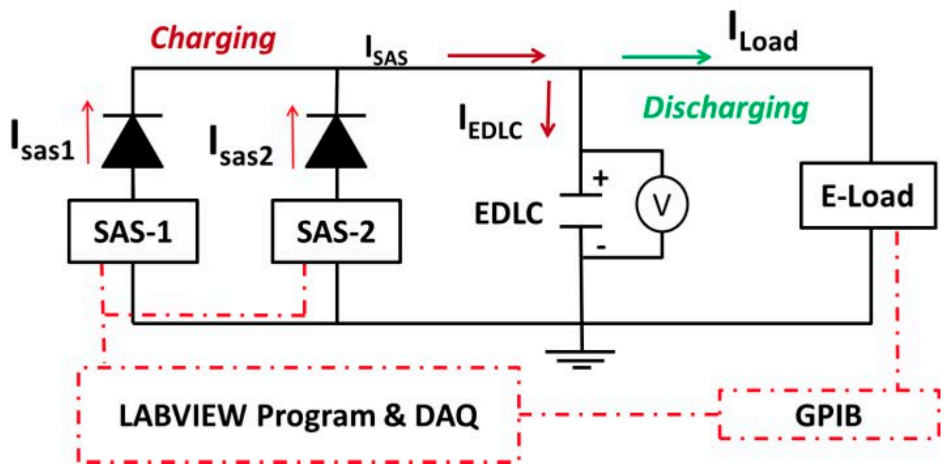

Figure 4:5 The schematic of the test for Case-1

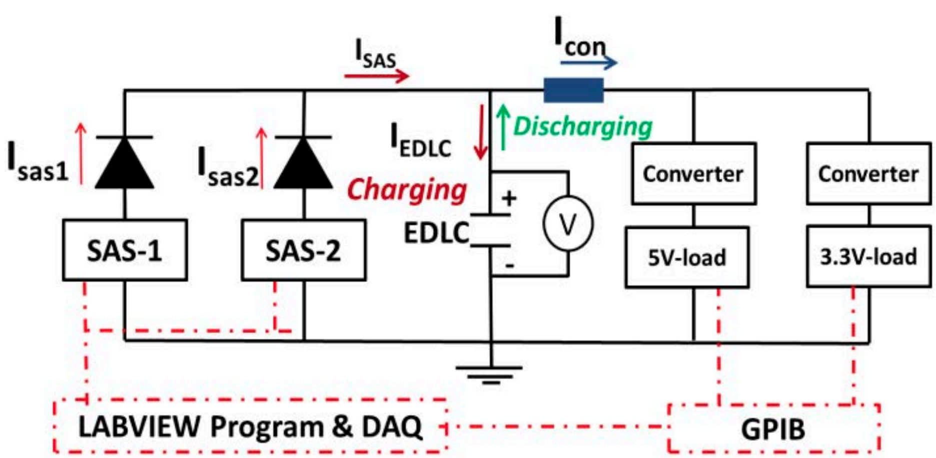

Figure 4:6 the schematic of the test for Case-2

Each cycle has a period of 92 minutes, with 56 minutes of power being supplied from the two solar array simulators (SAS-1 and SAS-2) to charge the EDLC and onward delivery of power to the load and thereafter 36 minutes of eclipse time assigned for the EDLC to discharge to the load, while the solar array simulators were hibernated. The dotted lines and blocks shown in the schematics in Figure 4.5 are for the control connections. The operation is controlled using LabVIEW test simulation tool developed for this purpose.

The Case-1 was performed to characterize the potential power distributable to the load by the system while the Case-2 was intended to simulate the load condition based on the standard 3.3V/ 5V distribution bus system.

SAS power profile used for the simulation of the generated power supply due to solar cells for the mission is shown in Figure 4.7 below. This power profile assumes that a satellite is in the ISS orbit (388km altitude and 52 degree inclination. The case-1 solar power simulation was done with the basis that at least 30% of the body mounted cells were exposed to sun irradiation with the cells characteristics shown in Table 4.7.

Table 4.7 gives summary of the SAS simulated Voc, Vmp, Imp and Isc. They are different from the values shown in Table 4.4 as it was very difficult to simulate the small power by the SAS. This means we were simulating the worst condition when defect solar cells are used.

Table 4:7.Summary of different SAS inputted conditions

Mode	Vmp (V)	Voc(V)	Imp(A)	Isc (A)
Case-1	1.9	2.8	0.59	0.62
Case-2	1.9	2.7	0.791	0.8

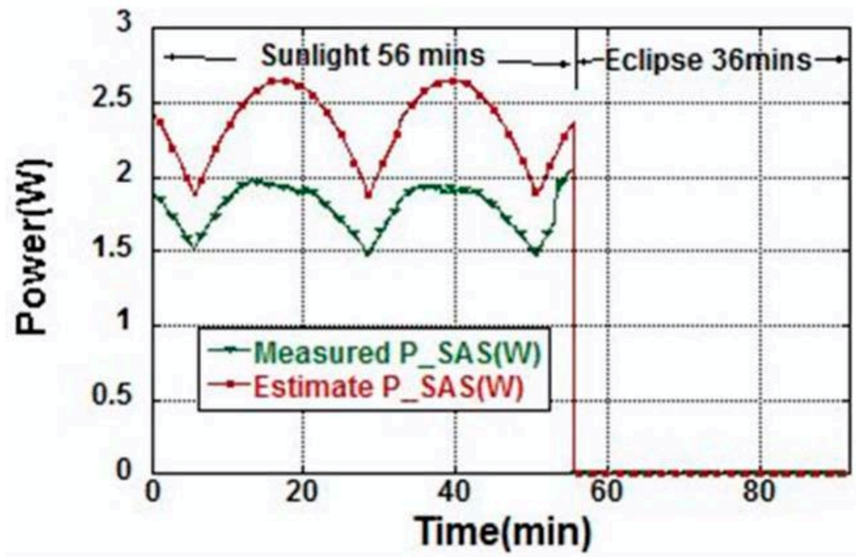

Figure 4:7 SAS power profile of Case-1

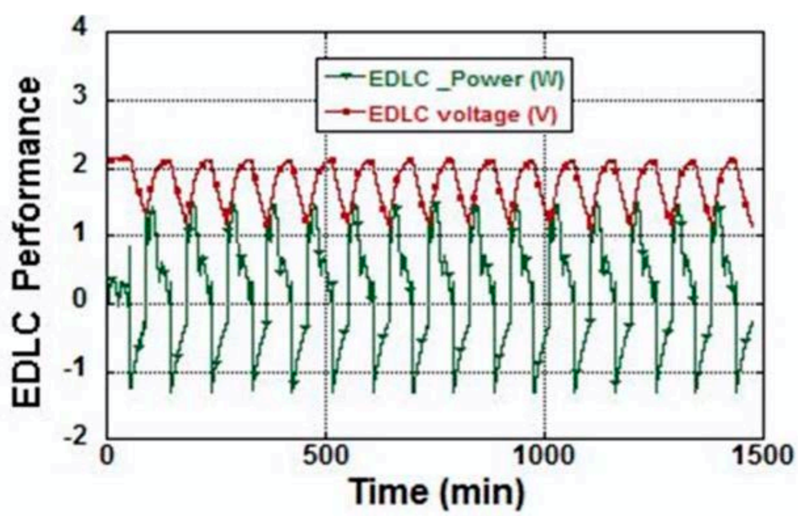

Figure 4:8 EDLC performance profile for one day (Case-1)

Figures 4.8 and 4.9 shows the graphs of the EDLC performance in the simulated orbit altitude of 388km for one day (16 cycles) and one cycle respectively for Case-1.
The discharge time (eclipse period) was 36 minutes and charging time (sunlight period) was 56 minutes. The load setting was at constant resistance (CR) of 2.8Ω so as to obtain as much as 1.3W (about +0.3W) margin for 1W power budget considering Vmp, 1.9V [$P=1.9V^2/2.8\Omega$]. At the end of discharge ,1.16V an output power of 0.5W [$P=1.16V^2/2.8\Omega$] was delivered [See the EDLC performance profile during one cycle in figure 4.9 and also load performance profiles in one cycle shown at Figure 4.10. For university CubeSat missions with power budget requirement of 0.5W to 1W this EDLC is very convenient; more so it is recoverable during sunlight after undergoing discharge as seen in the consistence of the performance for 16 cycles in one day[Alkali,2015].

Consistently, the end of charge voltage of the EDLC, V_{eoc} was 2.11V during sunlight periods of the simulation; during eclipse discharge time; the voltage had initial drop, V_{drop} 2.06Vdue to internal resistance. The instantaneous currents for end of charge,I_{eoc} and drop I_{drop} were 0.13A and -0.64A respectively. Within 36 minutes the end of discharge of the EDLC was 1.16V.

Notingly, the internal resistance ρ and charge q were calculated and shown in equations 4:13 & 4:14.

$$\rho = \frac{V_{eoc} - V_{drop}}{I_{eoc} - (-I_{drop})} = 0.065\Omega \qquad \text{Equation 4:13}$$

$$q = \int_{0}^{\Delta t} i\,dt = \text{- } 0.55\text{A} \times (36.2 \times 60)\text{s} = 1188\,C \qquad \text{Equation 4:14}$$

While the capacitance, C (Farad) is calculated/ verified using equation 4: 15 where q ($1188C$) and ΔV are the delivered charge and the difference in voltage from beginning of discharge after the instantaneous drop to end of discharge voltage (1.16V).The capacitance was unchanged even at change in voltage.

$$C = \frac{q}{\Delta V} = 1251F \qquad \text{Equation 4:15}$$

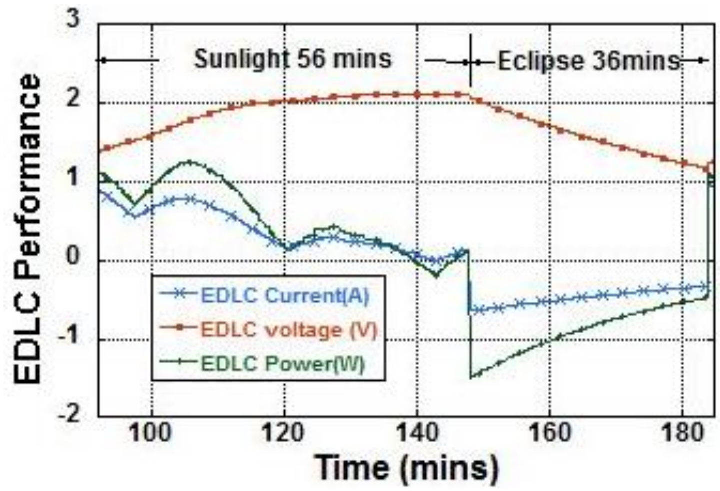

Figure 4:9 EDLC performance profile for 1 cycle (Case-1)

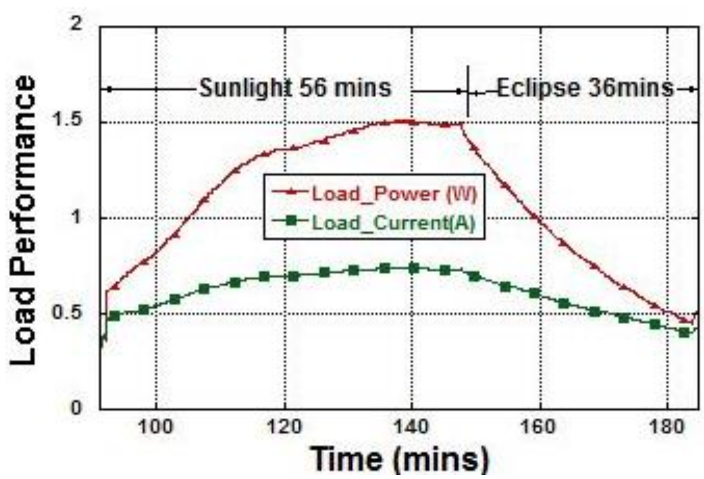

Figure 4:10 Load performance profile for 1 cycle (Case-1)

Figure 4.11 shows the expected power profile of the solar cells based on ISS inclined scenario.

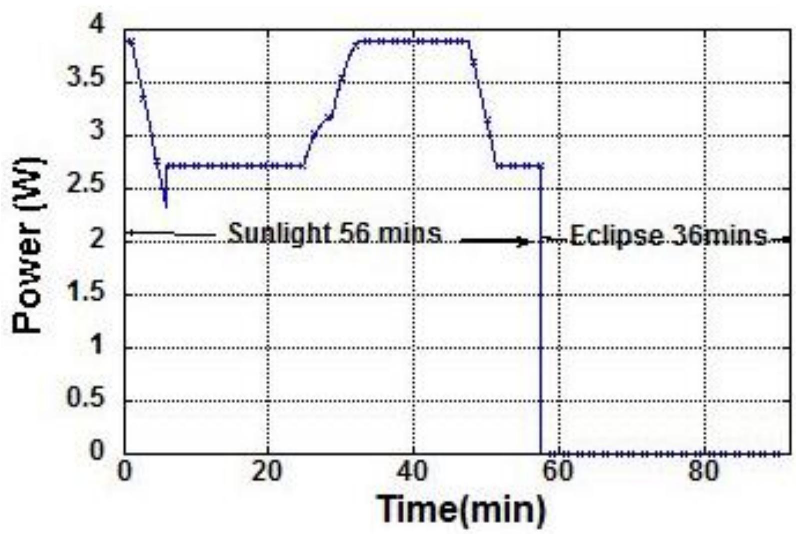

Figure 4:11 SAS power profile (ISS inclined case) [Alkali,2015]

For the second scenario (Case-2), the solar cell profile was based on the assumption that the Voc is 2.7.V (solar cell open circuit voltage available in the market) with voltage at maximum power (Vmp) 1.9V. The simulation was run for 17 cycles (to cover a day operation) with constant power of 0.65W supplied to the two electronic loads (3.3V and 5V buses).

Figure 4.12 shows the estimated and measured power profile of the SAS applied in Case-2. This condition simulates the polar earth orbit case with average generated power (2.04W) which is less than the average power (3.21W) from the actual ISS inclined scenario. This verified that at this worse condition the converters could effectively deliver to the 3V and 5V loads even at low EDLC voltage level. During sunlight, the point in the SAS power profile with zero power generation was the predicted time where the side with no cell was facing the sun and the last 36 minutes with zero power generation was due to the eclipse occurrence (solar cells were not radiated).

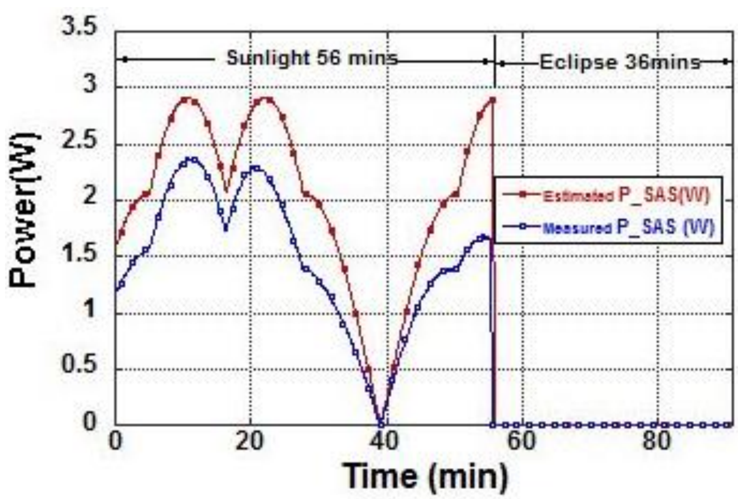

Figure 4:12 SAS power profile (Case-2) [Alkali,2015]

Figures 4.13 shows the EDLC voltage-power profile for 17 cycles (one day) and performance of the cell in one cycle is shown in figure 4.14.

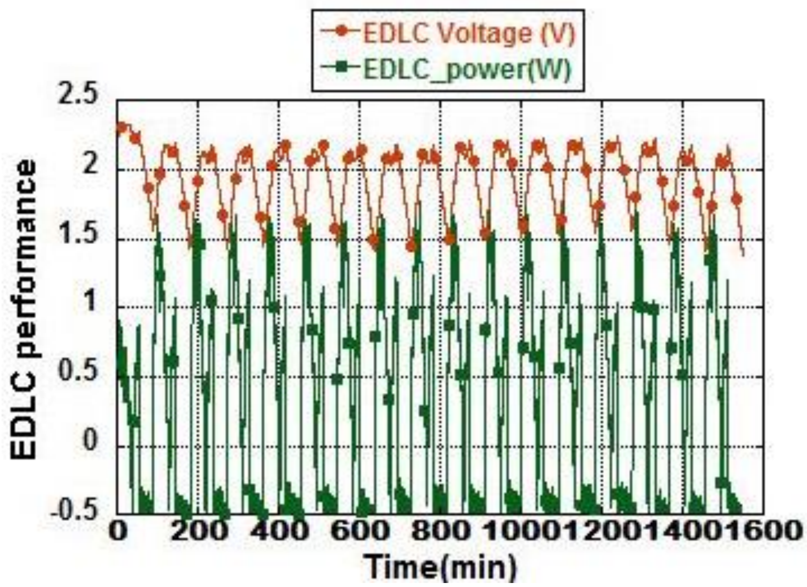

Figure 4:13 EDLC performance profile for 17 cycle (case 2)

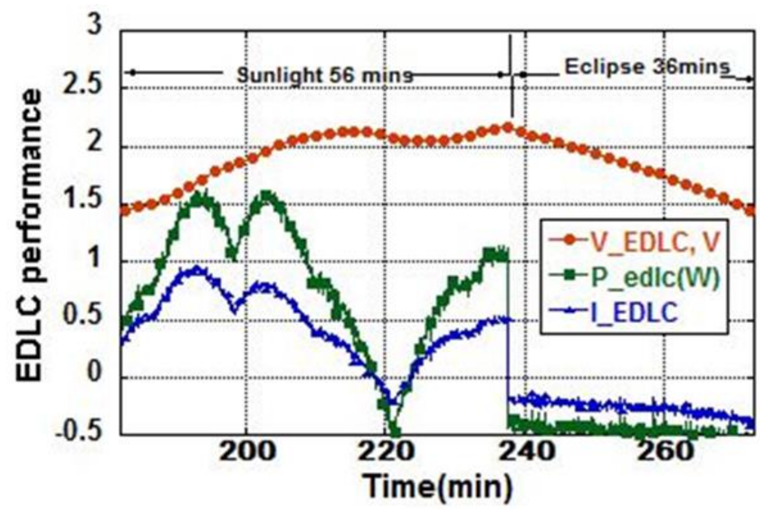

Figure 4:14 EDLC performance profile for 1 cycle (case 2) [Alkali, 2015]

For the load power profile under Case-2, 0.65W was constantly delivered for 24 hours where 0.34W went to the 5V bus and 0.31W was for 3.3V bus. During the eclipse time the power delivered to the load was from the EDLC. Figure 4.15 shows the power profiles with respect to time.

During one cycle, P_sas (i.e the power generated by the two solar arrays) delivered 2889J. On average it is 1.433W. The average was calculated by assuming the maximum power of the solar cell calculated from Figure 4.12, which was 2.02W. It is different from the actual SAS output, because SAS follows the I-V curve defined in Table 4.7, not necessarily operating at the peak power point. In comparing the estimated power produced by solar cell and the measured cells as shown in figure 4.15, the efficiency was 71%.The average power at the point before the DC/DC converter, P_con was 0.81W. The power consumed by the two loads (P_Load) was 0.65W. Therefore, the total efficiency is 80.2% comparing the load power to the SAS generated power, P_sas [Alkali, 2015].

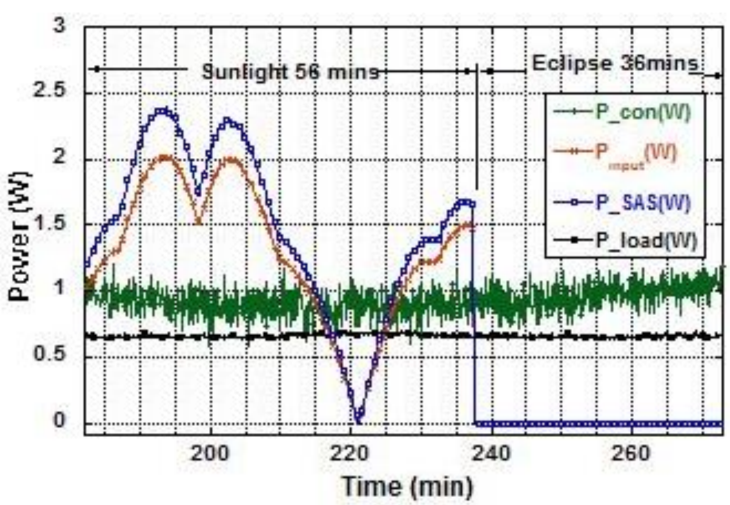

Figure 4:15 Load power profile for 1 cycle (case2)

Table 4.8 compares the state-of-art EDLC based EPS with the Ni-MH battery based EPS. The Ni-MH battery based EPS is chosen as it's more robust and safe compared to Lithium ion battery. The total volume of EPS including battery or EDLC is 178.6cm^3 and 85.75cm^3 for Ni-MH. Although EDLC volume consumes more than 17% of the total internal volume available for 1U CubeSat, its simplicity and robustness are attracting as the power system to be used for an educational satellite with limited power requirement.

Table 4:8Comparison of EDLC and battery based power systems[Alkali,2015]

Cell type	NiMH	EDLC
Mass (kg)	NiMH (x2) 0.0054	0.264
Storage unit (kg)	Battery +Box 0.14	0.264
Dimension	3.5cm x 3.5cm x 7cm (0.08575l) (with battery box)	7cm (L) x 5.7cm (ϕ) (0.179l)

4.4 Conclusions

After confirming the durability of Electric Double Layer capacitor (EDLC) under various harsh space environments in laboratory (Ref. 18), attempt has been taken to verify the performance as power source for Cubesats. Assuming and simulating very simple EPS topology for 1U CubeSat, performance of EDLC as the sole power storage device has been investigated when the average load power requirement is less than 0.65W even without a charging regulator. At two different power profiles, confirmed by solar array simulator (SAS), performance of EDLC has been checked and it passed satisfactorily. Although the energy density of the state-of-art EDLC is low compared to traditional secondary batteries but other featured plusses makes it's applicability as storage very attractive. On comparison of EDLC feasibility with traditional battery, currently in use, it is observed that the cumulative mass of the EDLC is higher but the ease of usage and simplicity of EDLC based power is more. In conclusion, EDLC can be used now and in future can get better with the energy density.

4.5 References

1. Alkali.M, Edries,M.Y., Khan,A.R., Cho,M., ''Laboratory verification of Electric Double Layer Capacitor based power system for a simple CubeSat mission'' International Journal of Electrical Energy(IjoEE) [Published September issue in September issues, volume 3,number 3, pp. 122-129].
2. Nanosatellites: Nanosats are go, Technology Quarterly Q2 2014 (the Economist). 2014-06-07. Retrieved 2014.09.5
3. http://iss.jaxa.jp/en/kiboexp/jssod/

4. M.Buchen,"2014 Nano/Microsatellite Market Assessment," SpaceWorks Enterprises, Inc. (SEI) Atlanta, GA (2014),http://www.sei.aero/eng/papers/uploads/archive/SpaceWorks_Nano_Microsatellite_Market_Assessment_January_2014.pdf
5. S.Nakasuka, R. Kawashima, "Micro/Nanosatellite Activities by Japanese Universities and Vision towards International Contribution" presented at Committee on the Peaceful Uses of Outer Space: 55th session 6-15 June 2012.
6. M.Alkali, T.Motohata, S.Tatsuo, H.Masui, M.Cho, "Supercapacitor: Testing it's practicability as power storage unit of a nanosatellite", presented at the 5th Nanosatellite symposium 20-22, November 2013, Tokyo.
7. M.Alkali, M.Y.Edries, A.R. Khan, H.Masui, M.Cho, "Preliminary Study of Electric Double Layer Capacitor as an Energy Storage of Simple Nanosatellite Power System", presented at 65th International Astronautic Congress (IAC), 29th September to 4th October 2014, Toronto, Canada.
8. Burke, A., "Ultracapacitors: why, how and where is the technology," *Journal of Power Sources*, Vol. 91, Issue 1, 2000, Pages 37-50
9. Conway, B. E., "*Electrochemical Supercapacitors Scientific Fundamentals and Technological Applications"*, Kluwer Academic/Plenum, New York, 1999, Chap. 2.
10. T.Sato,S.Marukane, T.Morinaga, K. Fukumoto, S. Yamazaki, "A thin layer including a carbon material improves the rate capability of an electric double layer capacitor," Journal of Power Sources
11. Fletchera, S.I., Sillars, F.B. , Carterb, R.C. , Crudenb, A.J., Mirzaeiana, M.,

Hudsonc, N.E. , Parkinsonc, J.A., Halla, P.J., "The effects of temperature on the performance of electrochemical double layer capacitors ," Journal of power sources Volume 195, Issue 21, 1 November 2010, Pages 7484–7488

12. T.shimizu,C.I.Underwood., "Power Subsystem Design for Micro-satellites Using supercapacitor Energy Storage", I presented at the 7th International Energy conversion Engineering Conference 2-5 August 2009, Denver, Colorado
13. M.Alkali., M.Y Edries,A.R Khan, H.Masui,M.Cho, "Environment Test Campaign of Commercial-off-the shelf Components of Electrical Double Layer Capacitor for space Use" presented at the,34th ISAS/JAXA Space Power Symposium ,6th March,2015, Institute of Space and Astronautical Science (ISAS)/Japan Aerospace Exploration Agency (JAXA) , Sagamihara, Japan
14. M.Alkali,M. Edries, H.Almubarak, A.R.Khan, H.Masui, M.Cho, "Performance evaluation of electric double layer capacitor as energy storage component of micro/nanosatellite on the imposition of varied temperature and vacuum conditions" , presented at the 50th AIAA/ASME/SAE/ASEE Joint Propulsion conference 28-30 July 2014, Cleveland Convention Centre, Cleveland ,Ohio
15. J.R Wertz,D.F Everett, J.F Puschell, "Space Mission Engineering: The New SMAD"
16. D.A. Vallado, W.D Mclain ,"Fundamentals of Astrodynanmics and Applications" 3rd Edition
17. M.Y. Edries, M. Alkali, M.Cho, "Design and evaluation of a Nano-satellite Battery Charge Regulator (BCR) based on a simple Maximum Power Point (MPP) tracking control", presented at the 58th Japan Society of Aeronautical and Sciences (JSASS)

conference, 12-14 November 2014, Nagasaki, Japan

18. M.Y. Edries, A.Tanaka,E.Dashdondog,H.O. Almubarak,M. Alkali,A.R.Khan,H.R Masui, M.Cho ,"Design and Testing of Electrical Power Subsystem (EPS) of a lean satellite, Horyu-IV" ,presented at the 30th ISTS ,34th IEPC & 6th NSAT Joint Conference, July 4th – 10th ,2015, Kobe-Hyogo, Japan
19. M.Alkali, M.Y.Edries, A.R. Khan, H.Masui, M.Cho, "Design Considerations and Ground Testing of Electrical Double Layer Capacitor as an Energy Storage Component for Nanosatellites", Journal of small satellite (JoSS) [published in October issues, Vol. 04, No. 02 (October 2015) pp. 387-405].

CHAPTER FIVE: GENERAL CONCLUSION AND FUTURE ISSUES

5.1 Conclusion

Electric Double layer capacitors are durable to harsh space environments such as wide operating temperature (thermal cycle test), launch environment (vibration test), separation (shock test), total ionization dose (TID of 20krad & 10krad in passive and active exposure respectively) and easily operable for simple mission.

The shortcomings for EDLCs in space applicability and applications over existing batteries include:

a. Low gravimetric energy density
b. Low volumetric energy density
c. Heritage
d. Low voltage level.
e. Non-space based Commercial off the shelf (COTS) components developed for terrestrial applications

Low gravimetric / volumetric energy densities of EDLC are quite a bottleneck in its utilization for EPS where dimension and mass are design consideration for most missions. In the nearest future this obstacle will be overcome with the current trend in capacitance, voltage and internal resistance enhancement. The ease of handling, power density and wide temperature operating range performance superiority relative to traditional batteries as against performance inferiority in terms of energy density are the consideration for trade-offs for applicability of EDLC now and near future as energy storage for spacecraft.

A simple EDLC based power system devoid of complexity with lessened COTS components for simple missions is considered. Also space environment experiments to characterize the COTS grade EDLC were performed to check survivability or durability of the cells for the existing harsh space environments cycle. The performance of the capacitor is investigated under harsh space environments such as wide operating temperature (thermal cycle test), launch environment (vibration test), separation (shock test) and total ionization dose. Judgement criteria were set to ascertain the feasibility of the EDLC cell as effective energy storage for space. The capacitor survived the durability test with degradation in capacitance (5%) and internal resistance (10%) which was less than the set criteria of 10% and 15% for capacitance and internal resistance respectively.

After confirming the durability of Electric Double Layer capacitor (EDLC) under various harsh space environments in laboratory, attempt has been taken to verify the performance as power source for CubeSat. Assuming and simulating very simple EPS topology for 1U CubeSat, performance of EDLC as the sole power storage device has been investigated when the average power requirement is less than 0.65W even without a charging regulator. At two different power profiles, confirmed by solar array simulator (SAS), performance of EDLC has been checked and it passed satisfactorily. Although the energy density of the SoA EDLC is low compared to traditional secondary batteries but other featured plusses makes it’s applicability as a storage unit very attractive. On comparison of EDLC feasibility with traditional battery, currently in use, it is observed that the cumulative mass of the EDLC is higher but the ease of usage and simplicity of EDLC based power is more. In conclusion, EDLC can be used now and in future can get better with the energy density. EDLC based power system is promising for commercial satellites in the nearest future.

5.2 Future Issues

In the present study, the longest charge –discharge cycle test was less than two weeks, far shorter than the typical mission duration of nanosatellites, six months at shortest. The long-life charging and discharging test with realistic power profiling under vacuum environment is a task of future work. Also there is a future plan to perform temperature cycle test for a period far longer than what was done in this studies.

Furthermore the usage of commercial spacecraft is highly achievable especially with the promising improvement in the energy density.

Spacecraft mass reduction especially with material enhancement such as application of carbon fibre re-enforced plastic (CFRP) in the structure can greatly enhance mass